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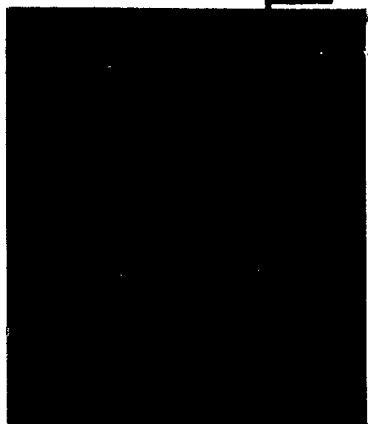
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NAI-56-304

RESEARCH AND REPORTS ON LAMINAR FLOW  
BOUNDARY LAYER CONTROL SYSTEMS

CONTRACT AF33(616)-3168  
TASK NO. 13618

PROGRESS REPORT FOR PERIOD  
1 MARCH THROUGH 31 MARCH 1956

REPORT TO WRIGHT AIR DEVELOPMENT CENTER

56 OORH 914-4

R.C. NO. 13190

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APPENDIX: INVESTIGATION OF THE FLOW IN A TUBE WITH LAMINAR SUCTION  
THROUGH 80 ROWS OF CLOSELY-SPACED HOLES, J. Goldsmith,  
March 1956, (BLC-86), Report No. NAI-56-293

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PROGRESS REPORT ON CONTRACT AF33(616)-3168

RESEARCH AND REPORTS ON LAMINAR FLOW BOUNDARY LAYER CONTROL SYSTEMS

I. BASIC CONTRACT PURPOSES

To supply the necessary personnel, services, and facilities to investigate laminar boundary layer control on wings and bodies through suction and to develop methods for the design and construction of a laminar boundary layer control airplane.

II. FACILITIES AND EQUIPMENT

There has been no change in the facility and equipment situation during the period covered by this report.

III. ORGANIZATION AND PERSONNEL

A. Organization

There have been no organizational changes during this period.

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B. Personnel

The staff now engaged full time on this contract consists of the following:

Engineering

Supervision	2
Clerical and Secretarial	2
Direct Charging	<u>24</u>
	28

Shop

Supervision	1
Clerical	1
Direct Charging	<u>17</u>
	19

Flight Test Department

Direct Charging (2 shifts) 9.5

IV. VISITORS

Dr. Fritz Haber, Consultant, Lycoming Division, Avco Corp.,  
Stratford, Connecticut

D. N. MacKay, West Coast Representative, Lycoming Division,  
Avco Corp., Beverly Hills, California

Major E. W. Geniesse, Hq., ARDC, USAF, Baltimore, Maryland

D. C. Hazen, Assistant Professor of Aeronautical Engineering,  
Princeton University, Princeton, New Jersey

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V. TECHNICAL PROGRESS

A. Theoretical Investigations of the Laminar Boundary Layer

1. Integration of Boundary Layer Equations over Tapered Swept Laminar Suction Wings in Incompressible Flow

Further work on these projects was deferred during the report period.

2. Integration of Boundary Layer Equations over Untapered Swept Laminar Suction Wings in Compressible Flow

The numerical integration for the problem under consideration (typical lower surface of a  $35^\circ$  swept laminar suction wing at 0.9 Mach number) was almost one-half completed.

3. Integration of Boundary Layer Equations over General Three-Dimensional Surfaces in Compressible Flow

The preparation of a report on a practicable method of solving this problem was continued.

4. Calculation of Stability of Incompressible Laminar Boundary Layers

This project was delayed by use of the personnel for another project. However, one trial of the proposed iterative method, incorporating a normalizing procedure analogous to that for eigenvalue problems in real variables, was made. As was originally suspected but not known, this trial indicated that a different and, in fact, quite dissimilar normalizing procedure is necessary in such problems in complex variables. Another trial with this different procedure is being started.

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5. Calculation of Stability of Compressible Boundary Layers

This project is being deferred pending the outcome of the trials of the iterative method on an incompressible problem.

6. Calculation of the Stability of Incompressible Laminar Boundary Layers under Crossflow Conditions

Some of the vortices generated on the rotating disc by three dimensional instability have been computed and plotted. They resemble those near the wall, as shown by Stuart\*. For our calculated distribution of disturbance velocities, we have been unable to find the other set of vortices far away from the wall which are shown in Stuart's figure.

Curves of constant excitation are being computed for the stagnation point profile on a swept wing and the vortex pattern will also be found then.

B. Basic Laminar Suction and Transition Investigations

1. Laminar Suction Experiments in the 2-Inch Tube with Suction through 80 Rows of Holes

This investigation has been completed and the results are presented in Report No. BLC-86.

2. Experiments in the 2-Inch Multi-section Tube

(a) Suction Experiments with Rows of Holes

As mentioned in last month's progress report, the data for 110 holes per row as measured for a revised test configuration were considerably differ-

\* Gregory, N., Stuart, J. T., and Walker, W.S.: On the Stability of Three-dimensional Boundary Layers with Application to the Flow Due to a Rotating Disk, Phil. Trans. of the Royal Society of London, No. 943, vol. 248, pp. 155-199.

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ent from data obtained from former measurements with 110 holes. Two changes were made for the revised configuration: (1) one of the holes was plugged and re-drilled since it was originally slightly misaligned with the other holes and (2) all the holes were reamed 0.001 inch oversize to insure uniform diameters. The improvement in measured critical suction quantities was greater than might be expected. As a result, it was felt that it would also be wise to repeat the measurements for other configurations after the holes were reamed. A small improvement was noted for 100 holes and no noticeable change occurred for 80 holes. Apparently, the improvement in suction distribution resulting from reaming the holes is very important for holes as closely spaced as 110 holes, but the importance of uniform circumferential suction becomes less pronounced as the hole spacing is increased.

(b) Experiments with a Single Roughness Element

No additional work was done in connection with these experiments, although preparation is being made to do experiments with distributed roughness.

(c) Suction Experiments with Chamfered Holes

The critical suction quantities for the unchamfered 80 holes have been measured, and they check with the results previously measured for 80 holes. Some special tools have been ground in order to chamfer the holes on the inside of the tube. A rig has also been designed which will hold a microscope and prism in the proper position for inspection of the chamfers.

(d) Pressure Drop through Suction Holes

The calculations on this project have been completed. It has been established that the measured suction losses through holes when the freestream

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velocity is zero, are considerably different from the measured losses when the freestream velocity is finite. Part of this difference in measured losses is due to the small, but finite, dynamic pressure in the boundary layer air which flows into the holes. When the suction air undergoes a rapid acceleration into the holes, the average dynamic pressure of this air accounts for most of the difference between measured losses with the wind on and wind off conditions. When the acceleration is small or negative, however, there is an additional loss which appears to be primarily a function of the contraction ratio of the air flowing into the holes. This work will be discussed in a report to be issued some time after the completion of the forthcoming wind tunnel program connected with the 142-inch body of revolution.

(e) Investigation of the Flow Phenomena at the End of a Suction Slot or Row of Holes

The calculation of additional parameters for this project have been started and are about 50% complete.

C. Wind Tunnel Tests on Two-Dimensional Wings

1. Swept Laminar Suction Wing Model

The design of the panel mentioned in previous reports is now 90% complete and is such that it may be installed in either the Northrop or University of Michigan wind tunnel with only minor modifications. Fabrication of the details and assemblies to be used in the panel structure is 55% complete. An instrument installation drawing has been released for fabrication.

2. 4%-Thick Straight Laminar Suction Wing Model

On thin wings necessary for supersonic flight, the pressure distribution over the forward portion of the airfoil approximates that of a flat plate. At

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high Reynolds numbers the laminar boundary layer of a flat plate is probably not sufficiently stable to attain transition Reynolds numbers corresponding to the point of minimum pressure on a practical wing. Therefore, on a thin wing, suction may have to be extended forward into the front region of the airfoil in order to stabilize the boundary layer. Since the rear pressure rise is small on a thin wing, relatively weak suction in this area will enable 100% laminar flow.

Suction experiments are also planned at small angles of attack in order to find out how to maintain laminar flow in the presence of the leading edge negative pressure peaks which occur at these angles of attack on thin wings.

A 4%-thick straight suction wing model is being designed for installation in either the 7-ft by 10-ft NAI tunnel or the University of Michigan tunnel. In order to achieve the highest Reynolds numbers possible, the panel under consideration will have a 17-ft chord which will allow wing chord Reynolds numbers of  $20 \times 10^6$  to  $27 \times 10^6$  in the Michigan tunnel. Relatively weak suction will be applied in the region of flat pressure from 0.05 c to 0.60 c, slightly stronger suction will be required in the region of the rear pressure rise. Relatively strong suction will be available in the region of the nose "suction peak" from 0.005 c to 0.10 c as it is needed. The suction quantities and the dimensions of the suction slots and holes located underneath the slots have been partially determined.

D. Wing Body Combinations with Laminar Flow

1. Experiment on a Flat-plate Wing Combination

A preliminary experiment was conducted on the flow in the intersection of a wing and a flat plate. The purpose of this investigation was to determine to what extent laminar flow could be maintained in and near the intersection of a wing-

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flat plate combination without the use of boundary layer suction.

The wing was constructed with an NACA 66-012 airfoil section of 24-inch chord. The flat plate had a chord of 30 inches and a span of 12 inches at either side of the wing. The model was so constructed that the leading edge of the airfoil coincided with that of the flat plate. In addition, the airfoil extended both above and below the plate so that the stagnation lines at the leading edges formed a symmetrical pattern.

The results of this experiment have not as yet been fully evaluated. However, a brief synopsis of the important findings will be given. It was possible to maintain laminar flow, in the intersection, for a distance of 10 inches aft of the leading edge for a free stream velocity of approximately 95 ft/sec. This corresponded to a length Reynolds number of 480,000. At a speed of 134 ft/sec the corresponding Reynolds number at transition was approximately 410,000. In order to achieve these results it was necessary to install a fillet in the intersection. The fillet was tapered from zero radius at the leading edge to 1.43 inches at 11 inches back from the leading edge. The fillet maintained constant radius from this point to the trailing edge.

The initial breakdown in the fillet spread to both the wing and the flat plate. The flat-plate breakdown was characteristic of that due to crossflow, that is, striations essentially parallel to the potential flow were observed. These striations extended out on the plate approximately 8 - 10 inches from the wing intersection. The transition line on the plate made an angle of approximately  $45^\circ$  with the main flow. The transition on the wing seemed to be that of a  $10^\circ$  turbulent wedge originating at the initial point of turbulence.

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For the case of no fillet, transition occurred first in the intersection six inches back from the leading edge, for a freestream velocity of 95 ft/sec. In addition, the flow on the wing near the intersection had present crossflow vortices, indicating incipient breakdown of the flow. Some of these vortices seemed to originate at or near to the leading edge of the wing.

A fillet of 3/8-inch radius was also tried but showed no improvement over the no fillet case.

Design studies for an optimum wing-plate configuration with different suction methods are continuing.

## 2. Wake Study of Wing-Body Intersection Problem

Work is being continued on the problem of preventing boundary layer breakdown aft of the intersection of a wing-body intersection. The feasibility of using a flat-sided airfoil is being investigated. This airfoil, consisting of parallel sides, would be mounted perpendicular to a flat plate and extended into the entrance nozzle of the wind tunnel.

The airfoil will be terminated in a suitable trailing edge contour to give a relatively high pressure recovery (see previous month's progress report). By these means, it is intended to prevent transition occurring due to difficulties at the leading edge intersection, and to reduce the crossflow on the plate adjacent to the wing ahead of the trailing edge pressure rise.

The velocity distribution for the above airfoil has been completed. The boundary layer development with suction over the pressure rise region at the trailing edge will be calculated.

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E. Bodies of Revolution

1. 96-Inch Elliptic Model

The 96-inch body with 20 slots between 5% and 22% of the body length is being prepared for the flight tests on the F-94 airplane. As a consequence of the wind tunnel measurements reported in last month's progress report, the slot width will be reduced to .003 inch.

2. 142-Inch Ellipsoid with Suction

The tests at the NAI wind tunnel are scheduled to begin on 9 April 1956.

3. Bodies of Fineness Ratio 9 for Transition Measurements

Since the transition on a body of revolution seems to depend primarily on the pressure distribution in the front part of the body, two more models are being built which have a more favorable pressure distribution than the ellipsoid. The first one is derived from the Sears Haack body  $(r = c [x(1-x)]^{3/4})$  where  $r$  is the radius of the body,  $x$  is the length coordinate,  $0 \leq x \leq 1$ , the second one from the parabolic body  $(r = c x (1-x))$ . In both cases, the contour was modified slightly to allow for a small nose radius. British calculations\* indicate that a body with a front part thinner than the parabolic does not maintain a uniform pressure distribution.

The design of the two models has been completed, both of which will be 96 inches long.

\* Neumark, S.: Velocity Distribution on Thin Bodies of Revolution at Zero Incidence in Incompressible Flow, R & M 2814, 1954.

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F. Aerodynamic Investigations of Suction Ducting Systems

A series of tests of the vee-inlet suction duct has been completed and the results are being analyzed. The tests investigated the losses through the duct system and the pressure distribution along the duct as a function of inlet flow rate ratio. The results show increasing energy losses as the percentage of flow through the high speed inlet is increased (both the inlet flow rate ratio  $Q_H/Q_{total}$  and the velocity ratio  $U_H/U_{duct\ exit}$  increase simultaneously, because the areas of the inlet turning-vane channels are fixed). An attempt will be made to determine how much of the energy loss is due to the kinetic energy loss inherent in mixing high velocity with low velocity streams; and how much of the loss is chargeable to the turning vanes, duct wall friction, and secondary flow effects.

G. F-94 Flight Laminar Suction Experiments

The flight tests on the effects of surface waviness have been continued. Measurements with the half wave of 2-inch length have been completed (crest wave). With this wave a maximum amplitude of .008-inch (waviness 1:250) was found to permit 100% laminar flow at all flight altitudes tested. The tests show that the flight speed has to be increased slightly with increasing waviness to provide a more favorable pressure gradient in the region of the wave.

A second crest wave of 6-inch wave length is under investigation. No laminar flow could be produced with an amplitude of .030 inch (waviness 1:200) under any flight condition. Reducing the waviness to nearly 1:250 (amplitude .023 inch) permitted 100% laminar flow at a few flight conditions with deflected flaps where a sufficiently steep favorable pressure gradient could partially compensate the detrimental effects of the wave. To provide additional data for a future analysis of the

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waviness experiments, static pressure pick-ups are being installed on the glove in the region of the wave.

#### H. Structural Investigations

##### 1. Full-Scale Wing Segment

The general equation for the buckling, under end compression and with all edges simply supported, of flat sandwich plates with any type of core was derived from Libove and Batdorf's Basic Differential Equation. (Ref. NACA Report 899, p. 6). The calculation of the critical compressive load per inch of width of panel, from the above general equation involves determining seven mechanical constants - two flexural stiffnesses, two Poisson Ratios, two shear stiffnesses and one twisting stiffness. A test jig, using two 4-inch x 20-inch sandwich specimens, has been designed to aid in the rapid evaluation of these constants.

The bonded screen assembly to provide simple supports for compression panels (outlined in Progress Report NAI-56-140, p. 12) proved feasible. However, a heavier screen gauge than that used in the tests (8 mils-20 mesh) is necessary to support the specimen at either end, where high shear loads develop. The bonding of the screen used to the unloaded edges of .15  $\pm$  thick plates was successful. No further investigation of the method is anticipated at present, as the structure of the panels proper is a matter of immediate concern.

The panel shown in the sketch was checked for two flexural stiffnesses [Longitudinal ( $D_x$ ) and Transverse ( $D_y$ )] and the transverse shear stiffness [ $D_{Qy}$ ] - all quantities per unit width.

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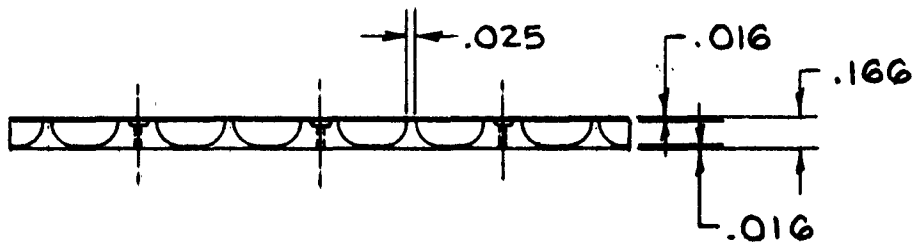
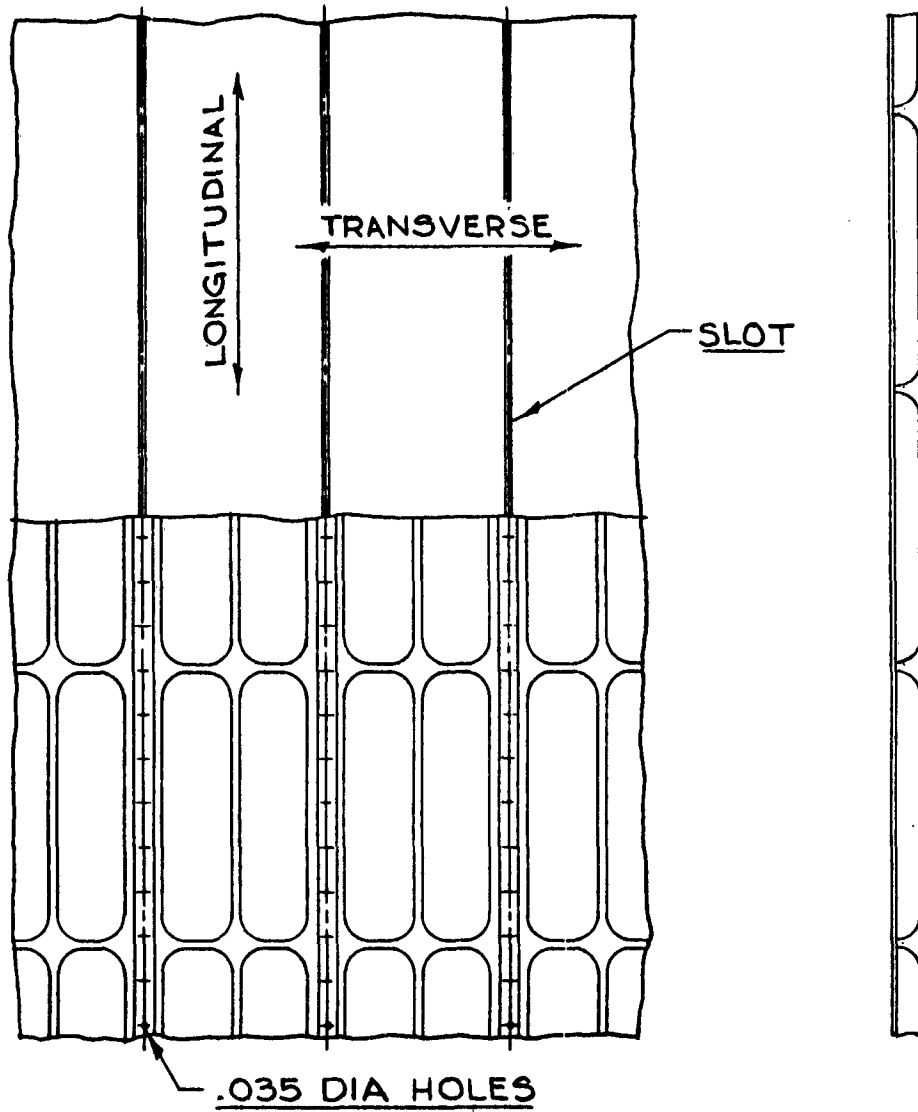
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	<u>D<sub>x</sub></u>	<u>D<sub>y</sub></u>	<u>D<sub>Qy</sub></u>
Unslotted		2000	800
Slotted	3100	715	200

A compressive buckling stress of 27,000 psi is anticipated.

The loss of stiffness of the slotted section is due to the shear deformation of the bond between the .025-inch feet and the slotted segments.

A group of panels with a variety of cores, calculated to be at the 50,000 psi buckling level, is being fabricated.

I. Design Studies of Hypothetical Long Range Laminar Suction Airplanes

1. Design Studies of Hypothetical Long Range Subsonic Supersonic Bombers with Low Drag Suction

Design studies on subsonic supersonic long range bomber type airplane with low drag suction were continued. A report describing the results of these studies will be submitted shortly.

2. Design Studies of a Hypothetical Long Range Laminar Suction Airplane with Supersonic Dash Performance

The aileron control problem on a 2.8 to 3.2%-thick wing has been studied. At subsonic speeds a small chord trailing edge midspan aileron provides ample rolling moment by virtue of the stiffness of the strut system. At supersonic speeds, however, the far aft center-of-pressure of the deflected aileron load and the small percentage of wing area effected by aileron deflection present a serious problem. A wider chord aileron was attempted without too much success. A calculation

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tion was made of the feasibility of using an outboard aileron operating intentionally above the reversal speed to generate rolling moment. Here again, the amount of rolling moment available is quite limited. One workable scheme of aileron control is to move the struts fore and/or aft at the strut body intersection, thereby twisting the wing panels. In this system the flap type ailerons would still be used for subsonic control and the strut actuation could be used for roll trim throughout the flight range. A spoiler aileron system is probably workable but not particularly compatible with the smoothness requirements of a laminar boundary layer airplane.

A preliminary two-dimensional type bending-torsion flutter investigation was made for the 3%-thick subsonic supersonic type airplane wing. The need of a small weight pod located outboard and forward of the wing is indicated to prevent bending-torsion flutter at high subsonic speeds.

3. Strut-Braced Wind Tunnel Model for the Wright Field  
10-Ft Transonic Model

The model has been completed and shipped to Wright Field along with the strain gage balance system in anticipation of a test date in the immediate future.

J. Powerplant Studies

1. Collection and Evaluation of Data on Existing Powerplants

Preliminary supersonic flight evaluation has continued for the series of turbofan engines proposed by Westinghouse and described in WADC Technical Report 55-239, parts 1, 2, and 3.

2. Subsonic Flight Performance Studies of Turbojet, Turbofan and  
Turboprop Engines Driving Suction Compressors through Special  
Turbine Stages

Basic comparisons of various open and closed-cycle types of engines

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has led to the selection of one closed-cycle engine for more detailed performance studies.

This engine is referred to as closed-cycle engine No. 12 and consists basically of the following components: A closed system comprising a compressor with one intercooler, a regenerator, a turbine driving a compressor supplying ram air to three burners, a turbine driving a ducted fan or a propeller, a turbine driving the closed-cycle compressor and finally an end cooler receiving the warm air from the regenerator. The suction compressor is driven by a turbine using the combustion gases leaving the burners. These turbine exhaust gases finally expand to the atmosphere through a nozzle and produce some thrust.

This system offers several advantages, such as relatively high efficiency, and small size combined with light weight and great flexibility of installation and component arrangement. On the other hand, it is quite complex and requires careful analysis and design.

To date, performance estimates for the closed-cycle engines have only been made for flight Mach No. of 0.9 at 70,000 feet using JP-4 fuel and air cycles with a limited range of cycle temperatures and pressure ratios. For these flight conditions a closed-cycle thermal efficiency of over 40% can be expected with conservative component efficiencies and over 50% with advanced component design and some variations from the arrangement in engine No. 12 described above. The detailed studies now starting on engine No. 12 will give a reasonably complete performance spectrum and realistic size and weight estimates for closed cycle components in general.

It is to be emphasized that variations in the fuel and the closed-cycle medium from JP-4 and air which have been used to date, will greatly affect both per-

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formance size and weight of these engines.

3. Preliminary Design of the Jet Exhaust Duct and Nozzle  
for a Suction Compressor Drive System Consisting of a  
Free Wheeling Turbine Stage

A 4-inch diameter duct model and related test facilities have been designed and are being fabricated.

4. Heat Transfer and Heat Exchanger Studies

Design studies on heat exchangers for the powerplants mentioned in (2) above have continued.

5. Single-Stage Axial-Flow Blower for Very Low Blade  
Chord Reynolds Numbers

Additional shop drawings of the torquemeter installation and the driving unit have been released.

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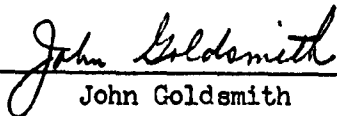
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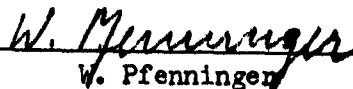
INVESTIGATION OF THE FLOW  
IN A TUBE WITH LAMINAR SUCTION THROUGH  
80 ROWS OF CLOSELY-SPACED HOLES

March 1956

**PREPARED BY**

  
John Goldsmith

**APPROVED BY**

  
W. Pfenniger

**REVISIONS**

CHG. NO.	DATE	ENGR.	PAGES AFFECTED	REMARKS

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## I. SUMMARY

Laminar suction experiments have been conducted in a 2-inch diameter low turbulence tube with 80 rows of closely-spaced suction holes at the downstream end of the tube. Laminar flow has been observed for these experiments for tube length Reynolds numbers from 8.5 to 18.8 million, and pressure rises in the suction region from 36 to 69% of the maximum dynamic pressure.

Profile drag coefficients of equivalent symmetrical suction airfoils having the same pressure distribution as the tube, have been estimated from the results of these experiments. The estimated drag coefficients for both wing surfaces at zero angle of attack (including the equivalent drag due to suction power) vary from .0007 to .0010 depending on the pressure rise and Reynolds number.

Similar experiments were previously conducted for 80 slots (Ref. 1), and a comparison is made between the performance of the slots and rows of holes. It was found that the slots are generally more efficient than the rows of holes, but not so much more efficient as to eliminate the use of rows of holes where other considerations may favor them.

## II. NOTATION

### General Subscripts

s	refers to reference station s in the tube
q	refers to reference station q in the tube
r	refers to reference station r in the tube
25	refers to reference station 25 in the tube

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General Subscripts (cont'd.)

- x refers to an arbitrary station in the tube or an arbitrary section in the diffuser
- o refers to the ambient atmosphere for the tube or equivalent airfoil
- t refers to the trailing edge of the equivalent airfoil

Linear Dimensions

- x 1) distance between the hypothetical straight tube station where the boundary layer thickness is zero and some downstream station in the 2-inch tube
- 2) distance from the leading edge of the equivalent airfoil along the chord line
- $x_D$  distance from the diffuser inlet to some downstream station in the diffuser
- $x_m$  value of  $x$  at the diffuser inlet =  $x_r + \Delta x_e$
- $\Delta x_e$  distance from station  $r$  to diffuser inlet
- $\Delta x_D$  length of diffuser
- $y$  radial distance from the inside tube wall-positive toward the tube centerline
- $z$  circumferential distance along tube wall
- $d$  diameter of suction hole

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Linear Dimensions (cont'd.)

- r radial distance from the tube centerline
- a tube radius
- $\bar{a}$  mean radius along the length of a diffuser section
- c chord length of the equivalent airfoil
- $\gamma$  centerline spacing between adjacent holes in a row
- D diameter of a suction flow measuring nozzle

Pressures

- S pressure differential across a suction flow measuring nozzle
- J jacket pressure referenced to static pressure at station r
- P static pressure in the tube referenced to atmospheric pressure, or at a station along the surface of the airfoil referenced to atmospheric pressure
- $\Delta H$  total pressure in the tube referenced to atmospheric pressure
- $\Delta P_D$  pressure at a point along the diffuser referenced to the static pressure at station r
- $\Delta P_q$  pressure differential in 2-inch tube inlet nozzle

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Pressures (cont'd.)

h total head pressure in the boundary layer at station 25 referenced to atmospheric pressure

q dynamic pressure

Suction Quantities

Q volume flow suction quantity at the suction holes

$Q_n$  volume flow suction quantity at the suction nozzles

$Q_c$  critical volume flow suction quantity for a row of holes

m mass flow suction quantity - slugs/sec.

Velocities

U velocity inside the tube

$\bar{U}$  mean velocity averaged over the cross-sectional area of the tube

$U_h$  velocity inside the tube at a row of holes

u velocity in the boundary layer at station 25

$U_B$  average centerline velocity averaged along the length of the tube to the diffuser inlet

$U_D$  average centerline velocity averaged along the length of the diffuser

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Velocities (cont'd.)

$V_o$  freestream velocity for the equivalent airfoil

$v$  velocity along the surface of the equivalent airfoil

$\bar{v}$  average velocity averaged along the chord of the equivalent airfoil

$v_m$  maximum velocity along the surface of the equivalent airfoil

Boundary layer parameters

$\theta$  momentum thickness at station 25 in the tube

$\theta_A$  momentum area at station 25 in the tube

$\theta_t$  momentum thickness at the airfoil trailing edge

$\delta^*$  displacement thickness of the boundary layer just ahead of a row of holes

$\delta$  total boundary layer thickness

Airfoil Coefficients

Note: 1) additional subscript  $x$  refers to a single suction chamber corresponding to a single diffuser section in the tube

2) additional subscript  $u$  refers to the upper surface only

$C_L$  two-dimensional lift coefficient

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Airfoil Coefficients (cont'd.)

- $C_D$  two-dimensional profile drag coefficient including equivalent pumping drag
- $C_{DQ}$  equivalent drag coefficient due to the suction power to accelerate the suction air to  $V_0$
- $C_{DW}$  wake drag coefficient
- $C_Q$  suction volume flow quantity coefficient
- $C_H$  pumping pressure rise coefficient

Miscellaneous

- $\mu$  viscosity coefficient of the air
- $\nu$  kinematic viscosity of the air
- $\rho$  density of the air
- $\rho_n$  density of the air at the suction flow measuring nozzle
- $\rho_j$  density of the air in the diffuser section suction chamber
- $\beta$  tube parameter  $x\nu / Ua^2$
- $\beta_m$   $\beta$  at the diffuser inlet
- $f$  minimum area of the suction flow measuring nozzle

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Miscellaneous (cont'd.)

F            maximum area of the suction flow measuring nozzle

$\alpha$            calibration factor for the flow measuring nozzle

$R_D$         Reynolds number of the flow measuring nozzle  
             based on the diameter and mean velocity

III. INTRODUCTION

By means of boundary layer suction, laminar flow can be maintained on an aerodynamic surface at high Reynolds numbers even in the presence of an adverse pressure gradient. The theoretical optimum type of suction appears to be continuous suction, or area suction, and some success has been achieved in maintaining laminar flow by means of suction through porous sheets. The use of porous sheets causes many practical difficulties, however, and it appears desirable to approach continuous suction in other ways.

Continuous suction has been approached by means of a large number of narrow slots, and it has been demonstrated that laminar flow can be maintained at high Reynolds numbers by means of such slots (see for instance Ref. 1 for tube experiments with 80 slots or Ref. 2 for flight tests with 12 slots). The boundary layer air removed at each slot must eventually pass through holes in an internal surface which is necessary to carry the structural loads across the slot. The requirement for this double skin structure does not appear to cause too much structural difficulty in flat or single curvature surfaces. In areas where there may be double curvature such as wing tips, or the juncture between the wing and fuselage, or engine nacelles, some difficulty may result in the construction of such a double

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skin. It appears, therefore, that suction through rows of holes in a single skin might be more practical for certain regions of an airplane. B. Carmichael was able to maintain laminar flow on the upper surface of a sailplane in flight by means of suction through holes for Reynolds numbers up to  $4 \times 10^6$  (Ref. 3).

In September 1953, experiments were begun in the Northrop Two-inch Laminar Flow Tube to determine whether or not extensive laminar flow could be maintained at high Reynolds numbers by means of suction through holes. In the first experiments suction was applied to a single hole in order to determine if disturbances would arise as a result of the 3-dimensional flow field in the vicinity of the single hole. It was observed that for suction quantities less than a given value (hereinafter denoted the critical suction quantity) suction through a single hole appeared to have negligible influence on the transition point. For suction in excess of the critical value, however, premature transition occurs. The results of these experiments are given in References 4 and 5. This critical suction quantity varies with the hole diameter and the boundary layer Reynolds number. The existence of this critical suction quantity does not mean that holes cannot be used for boundary layer suction in order to maintain laminar flow, but rather it establishes suction limits which cannot be exceeded if laminar flow is to be maintained.

Experiments were next conducted to determine whether or not there is interference between adjacent holes in a row. Many of these results are presented in Reference 5. For wide-spaced holes (perhaps fifty times the boundary layer displacement thickness) it was observed that each hole behaves as an individual isolated hole, and there is little change in the critical suction quantity per hole. For intermediate-spaced holes having centerline spacing of the order of

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ten times the boundary layer displacement thickness, it was observed that interference between adjacent holes was sufficient to considerably reduce the critical suction quantity per hole. For still closer spacing with the centerline distance of the order of 1.0 times the displacement thickness, a considerable change takes place in which there are two regions of laminar flow. When the suction quantity exceeds the lowest critical the flow becomes turbulent, but when still more suction is applied there is a second laminar region. A typical curve of critical suction quantity versus centerline spacing is shown in Figure 4 for such a configuration. It should be noted that below a certain centerline spacing the low level suction turbulent region does not exist, and there is a single very high critical suction quantity. For the reader who is interested in pursuing the subject, the critical curve shown in Figure 4 is very nicely explained in Ref. 10, in which smoke observations of the flow pattern between adjacent holes are described.

It remained to be determined whether or not there is interference between holes in adjacent rows of holes. The results of these experiments are given in References 5, 6 and 7. To summarize, the critical suction quantity per row of holes is generally reduced as additional rows of wide and intermediate-spaced holes are added in the tube. For a few configurations this reduction was not observed, but the associated flow pattern is rather complicated, and not yet fully understood. For very close spacing of the holes such as 110 holes around the circumference of the 2-inch tube, the addition of more than one row of holes did not appear to reduce the critical suction quantity when the suction was properly adjusted for each row of holes (Ref. 7). Generally, the behavior of

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10 rows with 110 holes per row in the 2-inch tube appeared to be approximately the same as for 10 slots. As a result, a diffuser was built with 80 rows of closely-spaced holes and the experiments similar to those described in Ref. 1 were repeated using rows of holes instead of 80 slots. It is with these experiments that this report is concerned.

#### IV. EXPERIMENTAL EQUIPMENT

In order to have a nearly direct comparison between the performance of slots and rows of closely-spaced holes, the experimental arrangement for 80 rows of holes duplicates, as nearly as possible, an earlier test configuration for 80 slots. In some ways, however, it was either impossible or impractical to make the two configurations exactly comparable, and the ways in which they differ are mentioned along with the description of the test setup for the 80 rows of holes given below. A complete description of the 80-slot configuration is given in References 1 and 8.

In all of these experiments a 2-inch i.d. tube provides a laminar boundary layer at high Reynolds number and low turbulence. Due to the increase in boundary layer displacement thickness as the air flows downstream in the tube, there is an effective reduction in cross-sectional area, and a slightly favorable pressure gradient results. The pressure distribution in the tube, therefore, is not unlike the pressure distribution on the fore part of a laminar flow airfoil ahead of the peak velocity point. The experimental setup for this basic tube is shown in Figure 1 and is described more fully in Ref. 9. The calibration of the basic tube parameters is also given in Ref. 9, and it is shown that these parameters are all functions of the single parameter  $x\sqrt{\nu} / \bar{U}a^2$ , which is easily determined as des-

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cribed in a later section. The tube is always operated in such a manner that the boundary layer thickness is considerably less than the radius of the tube; i.e., the total head along the length of the tube centerline is constant for any given run.

The boundary layer suction "diffuser" is installed at the downstream end of the basic tube, and is divided into eight separate sections containing 10 rows of holes (Figures 1 and 2). The inside diameter of this "diffuser" tapers slightly so that actually the air experiences a slight contraction when there is no suction in the holes or slots. With suction, however, the removal of air more than offsets the effect of the taper, and an increase in pressure results; hence the name "diffuser". For the 80 slot configuration the diameter variations occurred at each slot since the surface between each slot could be machined separately. For 80 rows of holes, however, a step at the row of holes was not desirable, and each section containing 10 rows was machined with a uniform taper. The variation of diameter along the diffuser for 80 rows of holes is given in Figure 3 together with the variation for 80 slots.

The streamwise distance between successive slots or rows of holes is generally 0.70 inches, except that for rows of holes the distance between adjacent rows at the section joints had to be increased to 1.168 inches in order to have room to seal the joints. As a result, the diffuser for the row of holes is about 7% longer than the slot diffuser.

Instrumentation is provided in each of the diffuser sections for measuring the suction pressure and airflow quantity for each group of 10 slots or rows of holes, and needle valves are provided for regulating these quantities. The airflow distribution through each slot in a single group of 10 is regulated by vari-

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ation in the depth of the slot, whereas the distribution of suction for the individual rows of holes in a group is regulated by means of orifices. In the latter case, a separate individual small chamber is provided for each row of holes, and the suction air passes from the small chamber into the large common chamber through orifices (Fig. 2) which can be opened or closed by means of cellulose tape.

Adjustment of the suction distribution is made as follows: Beginning with the upstream row of holes, the first group of orifices are adjusted (all other orifices sealed) to give the desired flow through the first row. Then, keeping the same chamber pressure, the second group of orifices are opened until the measured suction equals the desired total for the first two rows of holes. The flow quantity for the first row is not altered since the chamber pressure remains constant. This process is then continued for the third, fourth, etc., rows of holes until all eighty rows are adjusted. Of course, this adjustment of the suction distribution can be made for only one design condition, since the distribution will change somewhat when the chamber pressure or tube velocity is altered. The adjusted suction distribution and the corresponding design conditions are given in Figure 5.

The suction air in all eight chambers flows through the needle valves to a single suction box containing bleed holes and one or more sonic nozzles which lead to a compressor. Since in the diffuser, the pressure distribution depends on the suction distribution among the various eight sections, the pressure gradient along the diffuser can be changed by means of adjustment of the various needle valves. In this manner, experiments can be made for various total pressure rises and distributions of pressure. Pressure taps along the diffuser wall are provided

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for measuring this distribution.

A variable area sonic nozzle is also provided at the downstream end of the diffuser in order to vary the tube velocity and to reduce the compressor noise inside the tube.

#### V. MEASUREMENTS AND OBSERVATIONS

In order to reduce experimental test time to a minimum, full use was made of the results of the initial 2-inch tube experiments presented in Ref. 9. In particular, the results of these experiments confirm the theory that the velocity and state of the boundary layer are unique functions of the parameter  $x\sqrt{\nu}/U_a^2$ . It is only necessary, therefore, to measure the velocity at a single station in order to know the complete state of the flow in the tube as far downstream as the first row of suction holes.

The following measurements and observations were made for each run:

- a. Ambient air temperature and pressure
- b. Pressure drop across the inlet nozzle of the 2-inch tube -  $\Delta P_q$
- c. Pressure at station s referenced to the atmosphere -  $P_s$
- d. Static pressure at station r referenced to the atmosphere -  $P_r$
- e. Static pressure rise distribution along the diffuser referenced to static pressure at station r -  $\Delta P_D$
- f. Suction chamber pressure relative to the static pressure at station r - J
- g. Pressure drop across the suction chamber flow measuring nozzles - S
- h. The dynamic pressure distribution through the boundary layer at the downstream end of the diffuser - (h -  $P_{25}$ )

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In addition, a stethoscope was used to determine the state of the boundary layer at the downstream end of the diffuser, and the experiments were limited to those in which the stethoscope indicated quiet non-turbulent airflow.

Meriam micromanometers were used to measure all pressures. The accuracy of the pressure measurements vary from about  $\pm .0005$  inch of water for small pressure increments to about  $\pm .002$  inch of water for large pressure increments as determined from repeated measurements of the same pressure over a period of time. Generally then, the accuracy of the results is better than a small fraction of 1%. One notable exception, however, is in the measurement of the dynamic pressure in the portions of the boundary layer close to the wall for the low Reynolds number experiments. For the measurement station closest to the wall, the error may be as high as 9%, but this error is reduced to only 3.5% in the calculation of  $\frac{u}{U_{25}} \left( 1 - \frac{u}{U_{25}} \right)$ , and the final error in calculation of  $\theta$  is of the order of 0.1%. The other notable exception is in the measurement of the suction quantities in the low Reynolds number experiments for small pressure rises. For these conditions, the pressure drop across the flow measuring nozzles may be in error by 2 or 3% so that the flow quantities will be in error by 1 to 1.5%. Some additional reduction in error is to be expected in the determination of the total suction quantity since the airflow is measured separately for each of ten separate chambers, and it is not likely that all errors will be in the same direction. At the higher Reynolds numbers and pressure rises, this error is reduced to less than 0.5%.

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## VI. EVALUATION OF THE TUBE DATA

### A. The Basic Tube Variables

$$U_q = \sqrt{\frac{2.0102 \Delta P_q}{\rho_q}}$$

where the constant includes the effect of area ratio in the nozzle. It is then a simple matter to evaluate the Reynolds number  $U_q a/\nu$ . The relation between  $U_q$  and  $\bar{U}$  has been calibrated in Reference 9 as a function of  $\bar{U}a/\nu$  and is easily converted to a function of  $U_q a/\nu$ . Knowing  $U_q$  it is a simple matter to determine  $\bar{U}$  and then  $x\nu/\bar{U}a^2$  for any station in the tube upstream of the suction holes. Once  $x\nu/\bar{U}a^2$  is known for any given station, all other pertinent parameters are also known from the calibration of the tube as given in Ref. 9.

The total pressure change through the screens at the tube centerline is  $\Delta H = P_s + q_s$  where  $q_s = .0051 \Delta P_q$ . The constant .0051 accounts for the area ratio between stations s and q. The dynamic pressure at station r is therefore

$$q_r = \Delta H - P_r$$

and

$$U_r = \sqrt{\frac{2q_r}{\rho_r}}$$

### B. The Diffuser Variables

The dimensionless pressure rise at any point in the diffuser is

$$\frac{\Delta P_D}{q_r} = \frac{P_x - P_r}{q_r}$$

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and the local dynamic pressure ratio is

$$\frac{q_x}{q_r} = \frac{U_x^2}{U_r^2} = 1.0 - \frac{\Delta P_D}{q_r}$$

from which  $U_x/U_r$  is readily found

The overall dimensionless pressure rise in the diffuser is of course

$$\left( \frac{\Delta P_D}{q_r} \right)_{25} = \frac{P_{25} - P_r}{q_r}$$

and also  $U_{25}/U_r = U_x/U_r$  at station 25

C. Definition and Evaluation of the Momentum Thickness at the Downstream End of the Diffuser

The momentum thickness is defined as the momentum area,  $\theta_A$ , divided by the local circumference of the tube.

$$\theta = \frac{\theta_A}{2 \pi a_{25}} = \int_0^{a_{25}} \frac{u}{U_{25}} \left( 1 - \frac{u}{U_{25}} \right) \frac{r}{a_{25}} dr$$

Since  $dr = dy$  and since the undisturbed stream at the tube centerline contributes nothing to the momentum thickness this can be written

$$\theta = a_{25} \int_0^{\delta/a_{25}} \frac{u}{U_{25}} \left( 1 - \frac{u}{U_{25}} \right) \left( 1.0 - \frac{y}{a_{25}} \right) \frac{dy}{a_{25}}$$

where

$$\frac{u}{U_{25}} = \sqrt{\frac{h - P_{25}}{\Delta H - P_{25}}}$$

This integration has been performed by "Simpsons Rule".

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#### D. Pumping Characteristics

The suction nozzles are similar in form to calibrated ones used in suction experiments at Zurich and the calibration curve is given in Ref. 1. For these nozzles the mass flow is equal to

$$m = \rho_n Q_n = \alpha \frac{f \sqrt{2}}{\sqrt{1-(f/F)^2}} \sqrt{\rho_n S}$$

where  $f$  is the minimum area of the nozzle  
 $F$  is the maximum area of the nozzle  
 $\alpha$  is the calibration factor

$\rho_n$  assumed equal to  $\rho_j$  since the pressure drop through the nozzle and screens is small

$\alpha$  is determined from  $R_D/\alpha$  and the calibration curve given in Ref. 1.

$$R_D/\alpha = \frac{D}{\mu} \frac{2\sqrt{\rho_n S}}{\sqrt{1-(f/F)^2}}$$

Since in the diffuser the diameter varies it is essential to determine the suction/unit circumference for each diffuser section. For this determination the mean radius along the length of each section is used.

$$\left(\frac{dQ}{dz}\right)_x = \frac{\rho_r m}{2\pi \bar{r}_x}$$

In order to convert the tube information to the equivalent airfoil (described in the next section) it is also necessary to determine the average velocity along the length of the tube. For the basic tube the value of  $U/U$

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can be written with good accuracy in the form

$$\frac{U}{\bar{U}} - 1 = 2.82 \beta^{.48}$$

where  $\beta = x \sqrt{U_a^2}$

The ratio of the average velocity to  $\bar{U}$  in the basic tube is therefore

$$\frac{U_B}{\bar{U}} = \frac{1}{\beta_m} \int_0^{\beta_m} (1 + 2.82 \beta^{.48}) d\beta = 1 + 1.9054 \beta_m$$

where  $\beta_m = \frac{x_m \sqrt{U_a^2}}{\bar{U}}$

For the diffuser

$$\frac{U}{U_r} = \sqrt{1 - \frac{\Delta P_D}{q_r}}$$

and the ratio of average velocity is

$$\frac{U_D}{U_r} = \frac{1}{\Delta x_D} \int_0^{\Delta x_D} \frac{U}{U_r} dx_D$$

and the integral has been evaluated by "Simpsons Rule".

Then 
$$\frac{U_D}{\bar{U}} = \frac{U_D}{U_r} \frac{U_r}{\bar{U}}$$

The overall average velocity is set equal to the average airfoil velocity  $\bar{V}$

so that

$$\frac{\bar{V}}{\bar{U}} = \frac{\frac{U_B}{\bar{U}} x_m + \frac{U_D}{\bar{U}} \Delta x_D}{x_{25}}$$

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## VII. THE EQUIVALENT AIRFOIL

If one imagines a symmetrical laminar suction airfoil with the same chordwise pressure distribution as measured along the length of the tube, it can be assumed (without too much error) that the boundary layer development along the airfoil is the same as for the tube. The profile drag of such an "equivalent" airfoil can be evaluated from the measurements made in the tube experiments.

In order to calculate the equivalent drag it is first necessary to estimate the equivalent freestream velocity  $V_0$  for the airfoil. Since the chordwise pressure distributions are similar to those for the NACA 6 Series airfoils, we may use the data of these airfoils (Ref. 11) as a guide to selecting a  $V_0$  and a corresponding trailing edge velocity. In Figure 6 the ratio of  $\bar{v}/V_0$  for several airfoils in the series is presented as a function of the thickness ratio where  $\bar{v}$  is the average velocity on the airfoil. It is seen that this ratio is very nearly a unique function of thickness. Since the average velocity in the tube is easily calculated, it can be surmized that  $V_0$  is fixed once a thickness ratio has been selected. The calculations for the equivalent airfoil in Ref. 1 use the value  $\bar{v}/V_0 = 1.06$  and the same ratio will be used in this report so that the results can be directly compared.

Figure 6 also shows the variation of  $v_t/v_{\max}$  as a function of the maximum velocity point on the airfoil. This curve can be used to select a diffuser pressure rise which corresponds to the trends shown by the NACA 6 Series airfoils, but it is necessary to extrapolate the NACA data to 85% chord since this corresponds to the tube experiments. The extrapolation is approximate, however, and

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the corresponding  $v_t/v_{\max}$  cannot be precisely determined, although it can be reduced to a reasonably narrow range. It is seen that for  $\nabla/V = 1.06$ , the ratio of  $v_t/v_{\max}$  is probably close to 0.72 at 85% chord, and this corresponds to 52% pressure rise. In order to avoid settling definitely on a single pressure rise, the calculations were made for all runs. This has the advantage that it shows the influence of the pressure rise on the airfoil drag.

Once a value of  $\nabla/V_o$  has been determined it is a simple matter to find  $V_o$  from  $\bar{U}$  since according to the foregoing assumptions the average velocity in the tube equals that on the airfoil

$$\left(\frac{\nabla}{\bar{U}}\right) / \left(\frac{\nabla}{V_o}\right) = \frac{V_o}{\bar{U}}$$

Also

$$\frac{v_m}{V_o} = \frac{U_r}{\bar{U}} \frac{\bar{U}}{V_o}$$

$$\frac{v_t}{V_o} = \frac{U_{25}}{U_r} \frac{v_m}{V_o}$$

$$c = x_t$$

The net pressure to be supplied by the suction pump is defined as the difference between the jacket pressure and the freestream total ( $q_o$ ), and in coefficient form this becomes for each section " $x$ ";

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$$C_{Hx} = \frac{q_r - J_x}{q_r} \frac{q_r}{q_o} = \left(1 - \frac{J_x}{q_r}\right) \left(\frac{v_m}{V_o}\right)^2$$

The suction quantity volume flow coefficient for each section is

$$C_{Qx} = \frac{dQ/dz}{V_o c}$$

and the equivalent drag coefficient for each section is

$$C_{DQx} = C_{Qx} C_{Hx}$$

Then

$$\left. \begin{aligned} C_{DQu} &= \sum_0^8 C_{DQx} \\ C_{Qu} &= \sum_0^8 C_{Qx} \end{aligned} \right\} \text{ where the } u \text{ denotes upper surface only}$$

The wake drag is determined from the measured momentum thickness at station 25. According to R & M 1838 (Ref. 12)

$$C_{DWu} = \frac{2\theta_t}{c} \left(\frac{v_t}{V_o}\right)^{3.2} \text{ where } \theta_t = \theta$$

The total drag for one surface is, therefore

$$C_{Du} = C_{DWu} + C_{DQu}$$

And if the airfoil is symmetrical and at zero angle of attack

$$C_D = 2C_{Du}$$

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### VIII. DISCUSSION

In a tube, where the boundary layer thickness is a finite proportion of the tube radius, there is a direct relation between the suction distribution and the pressure distribution. The data and curves presented in this report do not represent, therefore, any particular optimum operating conditions, but rather they simply represent conditions under which laminar flow persists even in the presence of adverse pressure gradients. Data obtained in the experiments are tabulated in Table I, a-c for a diffuser pressure rise range of about 36% to about 69% and over a Reynolds number ( $\bar{U}x/\nu$ ) range of about  $8.8 \times 10^6$  to the maximum for which laminar flow could be maintained. For the smallest pressure rises it will be noted that experiments could be made up to a Reynolds number of about  $18.8 \times 10^6$ , but as the diffuser pressure rise was increased the upper Reynolds number for laminar flow was reduced. For a pressure rise of 69% the maximum Reynolds number for laminar flow in these experiments was about  $14.0 \times 10^6$ . Again, it should be pointed out that these limits are a function of the interrelationship between pressure rise and suction distribution in the tube experiments and do not necessarily represent limits for laminar flow. If the suction could be increased without also increasing the pressure rise, the upper limits for laminar flow might be quite different. Laminar flow can also be maintained at pressure rises less than 36%, but for good efficiency it would be necessary to readjust the design suction distribution from the one given in Figure 5 to one having a much lower pressure rise. This readjustment would be necessary in order to maintain a more uniform suction along the length of each separate diffuser section. Figures

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7 through 9 show typical measured results for the tube experiments. In particular, it should be noted in Figures 7 and 8 that both the Reynolds number and the overall pressure rise have considerable influence on the value of  $\theta$  at the diffuser exit. On the other hand, the distribution of pressure (or suction) along the length of the diffuser appears to have a negligible influence on the exit profile as seen in Figure 9. The profile represented by circles in Figure 9 is out of line more because of the difference in Reynolds number than because of the pressure distribution.

The boundary layer profiles shown in Figures 7 through 9 are laminar profiles even though they may not have the shape generally associated with laminar profiles. The unusual shape is the result of the combination of boundary layer suction and the adverse pressure gradient.

A comparison of the efficiency of the suction system could be made for the tube itself for each configuration, but it is believed that such a comparison has more meaning when made for the equivalent airfoils. Figure 10A shows the 2-dimensional airfoil coefficients of wake drag and profile drag (including effective pumping drag) for each of the experiments with linear pressure rise. The equivalent freestream velocity was determined by assuming  $\nabla/V_0 = 1.06$  as discussed in the previous section so as to be consistent with Ref. 1. From Figure 6 it is seen that the parameter  $v_t/v_{\max}$  would be expected to be about 0.72 for  $v_{\max}$  at 85% chord (corresponds to tube experiments). The corresponding pressure rise is about 52% of the maximum dynamic pressure, so that the diamonds in Fig. 10 would most closely represent an airfoil which follows the trends of the NACA 6 series.

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The dashed line in Figure 10A is the drag coefficient of a theoretical laminar flat plate (both sides exposed to the airstream). The wake drag coefficient of the equivalent airfoil is about .0002. When the equivalent suction drag is included for the airfoil with 52% pressure rise the drag is in excess of the flat plate drag by about 15% at a Reynolds number of  $17 \times 10^6$  and by about 7% at a Reynolds number of  $10 \times 10^6$ .

The volume flow coefficients do not mean too much in these experiments, since the volume flow was set at the value necessary to produce the desired pressure distribution and do not represent any sort of optimum value. They are presented in Fig. 10B, however, for those who may be interested in their magnitude. For these experiments, the coefficient is very nearly a unique function of the total diffuser pressure rise as might be expected from the relation between suction and pressure distribution in a tube. The flow has always been laminar for the experiments, however, and one can conclude that these suction quantities are adequate. From flight tests on a laminar flow glove, it can also be concluded that the flow coefficients in Figure 10B are not overly large, since flight experiments with slots indicate required flow coefficients of the order of  $3.0$  to  $6.0 \times 10^{-4}$  for one surface only.

The influence of trailing edge pressure distribution is shown in Fig. 11. The wake drag is effected very little as would be expected from the study of Fig. 9. Some small variation results in the equivalent drag due to the suction power, however, because of the fact that the required suction quantities are reduced for the upstream suction chambers where the required pumping pressure differential is greatest. Compensation for this deficit in suction quantity occurs

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in the downstream chambers where the required pumping pressure differential is smaller; consequently the pumping power is less.

A rough comparison between the results for 80 slots and 80 rows of holes with a 43% pressure rise is presented in Figure 12. It is seen that generally the experimental conditions are comparable with two exceptions. The experiment with the holes is for a somewhat higher Reynolds number, since this experiment was made on a rather cold day having a low value of  $\nu$ . This difference should be favorable to the holes judging from the results given in Figure 10. On the other hand, the diffuser pressure distribution is more curved (more convex) for the slots, and according to Fig. 9, this difference should make the slots appear slightly more favorable. The greatest difference in measured results shows up in the diffuser exit profile indicating that the rows of holes are not quite as efficient as the slots for boundary layer control. Still, the overall drag coefficient of  $C_D = .000707$  for the holes is only about 3.5% greater than for the continuous slots. If Fig. 11 is used to estimate a correction in the drag coefficient to a Reynolds number of  $15.6 \times 10^6$  (comparable to the slots), then the total drag coefficient would be about .000715 or about 4.5% greater than for slots.

The results presented in Fig. 12 are for the only two experiments in which all operating conditions are approximately equivalent for both slots and rows of holes. If we disregard the influence of the diffuser pressure distribution, a comparison can be made over the Reynolds number range of the experiments presented in Table III of Ref. 1. Some of these results are re-tabulated in Table II of this report. These experiments with slots have convex pressure distributions

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similar to the one shown in Fig. 12. Comparable results for rows of holes with linear pressure distributions are also given in Table II. For this comparison the difference in drag coefficient varies from about 9% to 14% for Reynolds numbers of  $15.6$  to  $20.0 \times 10^6$ , respectively.

It can be concluded from the foregoing discussion that the continuous slots are somewhat more efficient than rows of holes, yet they are not so much superior that rows of holes should be completely ruled out. From Figure 10, the sectional drag coefficients average approximately .0008, and this is still far superior to drag coefficients of .0040 as measured for the best conventional airfoils. For surfaces of complex curvature, and for certain experimental work it appears that rows of holes may have a definite application.

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# REFERENCES

- 1 Pfenninger, W., Moness, E. and Sipe, O. E.: Investigation of Laminar Flow in a Tube at High Reynolds Numbers and Low Turbulence with Boundary Layer Suction Through 80 Slots, Northrop Aircraft, Inc., Report No. BLC-53, July 1954.
- 2 Pfenninger, W., Groth, E. E., Carmichael, B. H. and Whites, R. C.: Low Drag Boundary Layer Suction Experiments in Flight on the Wing Glove of an F94-A Airplane. Phase I - Suction Through Twelve Slots (Includes Appendices I, II and III), Northrop Aircraft, Inc., Report No. BLC-77, April 1955.
- 3 Carmichael, B. H.: Flight Observations of Suction Stabilized Boundary Layers, Eng. Res. Station of Miss. State College, Research Report No. 4, January 1953.
- 4 Meyer, W. A., Goldsmith, John and Pfenninger, W.: Note on Preliminary Laminar Suction Experiments through Round Holes at High Reynolds Numbers and Low Turbulence, Northrop Aircraft, Inc., Report No. BLC-12, September 1953.
- 5 Goldsmith, John and Meyer, W. A.: Preliminary Experiments on Laminar Boundary Layer Suction through Circular Holes at High Reynolds Numbers and Low Turbulence, Northrop Aircraft, Inc., Report No. BLC-23, November 1953.
- 6 Goldsmith, John: Additional Experiments on Laminar Boundary Layer Suction through Circular Holes at High Reynolds Numbers and Low Turbulence, Northrop Aircraft, Inc., Report No. BLC-28, February 1954.
- 7 Goldsmith, John: Experiments with Laminar Boundary Layer Suction Through Rows of Closely Spaced Circular Holes at High Reynolds No. and Low Turbulence, Northrop Report No. BLC-36, March 1954.
- 8 Meyer, W. A. and Pfenninger, W.: Preliminary Investigations of Laminar Flow in a Tube at High Reynolds Numbers and Low Turbulence with Boundary Layer Suction through 80 Slots, Northrop Aircraft, Inc., Report No. BLC-7, August 1953.
- 9 Pfenninger, W.: Experiments with Laminar Flow in the Inlet Length of a Tube at High Reynolds Numbers with and Without Boundary Layer Suction, Northrop Aircraft, Inc., May 1952.

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REFERENCES (CONT'D.)

- 10 Meyer, W. A.: Preliminary Report on the Flow Field Due to Laminar Suction Through Holes, Northrop Aircraft, Inc., Report No. BLC-75, March 1955
- 11 Abbott, Ira H., von Doenhoff, Albert E., and Stivers, Louis S., Jr.: Summary of Airfoil Data, NACA Report No. 824, 1945.
- 12 Squire, H. B. and Young, A. D.: The Calculation of the Profile Drag of Airfoils, British R & M 1838, November 1937.

ENGINEER J. Goldsmith		<div style="text-align: center;"> <b>CONFIDENTIAL</b>  <b>NORTHROP AIRCRAFT, INC.</b>  <b>EXPERIMENTAL DATA</b>  <b>TABLE I-a</b> </div>										PAGE 29	
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$(\bar{U} \times \bar{V}) / 10^{-4}$	8.786	11.006	14.269	17.047	18.798	8.492	10.906	13.867	17.057	18.799			
$(\bar{U} \times \bar{V}) / 10^{-4}$	10.340	12.684	16.065	18.918	20.700	9.993	12.530	15.605	18.850	20.626			
$(\bar{U} \times \bar{V}) / 10^{-4}$	3.554	3.570	3.588	3.599	3.625	4.337	4.220	4.362	4.405	4.412			
$\bar{U} - \bar{V} / \bar{V}$	44.04	56.56	72.52	88.59	97.66	44.72	56.73	73.18	88.57	97.65			
$\bar{U} - \bar{V} / \bar{V}$	59.94	74.87	93.18	111.93	122.41	61.02	75.08	94.52	111.87	122.22			
$(\bar{U} \times \bar{V}) / 10^{-4}$	1.627	1.669	1.650	1.687	1.687	1.709	1.689	1.7132	1.6859	1.6864			
$(\bar{U} \times \bar{V}) / 10^{-4}$	2.337	2.296	2.318	2.280	2.281	2.266	2.281	2.261	2.282	2.281			
$\theta - \omega$	0.05112	0.04238	0.03759	0.03424	0.03425	0.03444	0.04336	0.03741	0.03353	0.03106			
$Q - \bar{V} / \bar{V}$													
SECTION 1	0.01956	0.02483	0.02735	0.03010	0.03239	0.02385	0.02589	0.03048	0.03598	0.03875			
2	0.01821	0.02260	0.02750	0.03402	0.03715	0.02347	0.02691	0.03363	0.04021	0.04253			
3	0.01954	0.02720	0.03037	0.03665	0.03869	0.02494	0.02853	0.03775	0.04290	0.04743			
4	0.02202	0.02654	0.03068	0.03882	0.04161	0.02735	0.03068	0.03826	0.04586	0.05064			
5	0.02241	0.02754	0.03462	0.04205	0.04499	0.02769	0.03354	0.04414	0.05078	0.05612			
6	0.02120	0.02818	0.03127	0.04004	0.04379	0.02634	0.03099	0.04224	0.05096	0.05700			
7	0.02204	0.02880	0.03586	0.04545	0.04766	0.02987	0.03714	0.04856	0.05549	0.06143			
8	0.02165	0.02913	0.03632	0.04516	0.04992	0.02987	0.03409	0.04701	0.05843	0.06300			
1-(7/8)													
SECTION 1	1.0347	1.0234	1.0114	1.0409	1.0100	1.0444	1.0289	1.0216	1.0189	1.0174			
2	9.827	9.767	9.730	9.749	9.711	9.840	9.806	9.763	9.720	9.674			
3	9.819	9.878	9.816	9.872	9.849	9.898	9.864	9.833	9.855	9.814			
4	9.898	9.837	9.883	9.884	9.896	9.882	9.815	9.816	9.877	9.891			
5	9.827	9.877	9.882	9.870	9.861	9.870	9.815	9.817	9.860	9.866			
6	7.858	7.852	7.828	7.816	7.822	7.400	7.470	7.417	7.397	7.410			
7	7.400	7.367	7.387	7.365	7.357	6.858	6.961	6.877	6.822	6.827			
8	6.941	6.898	6.905	6.867	6.895	6.291	6.322	6.244	6.265	6.260			
Notes-Data	LINEAR	LINEAR											





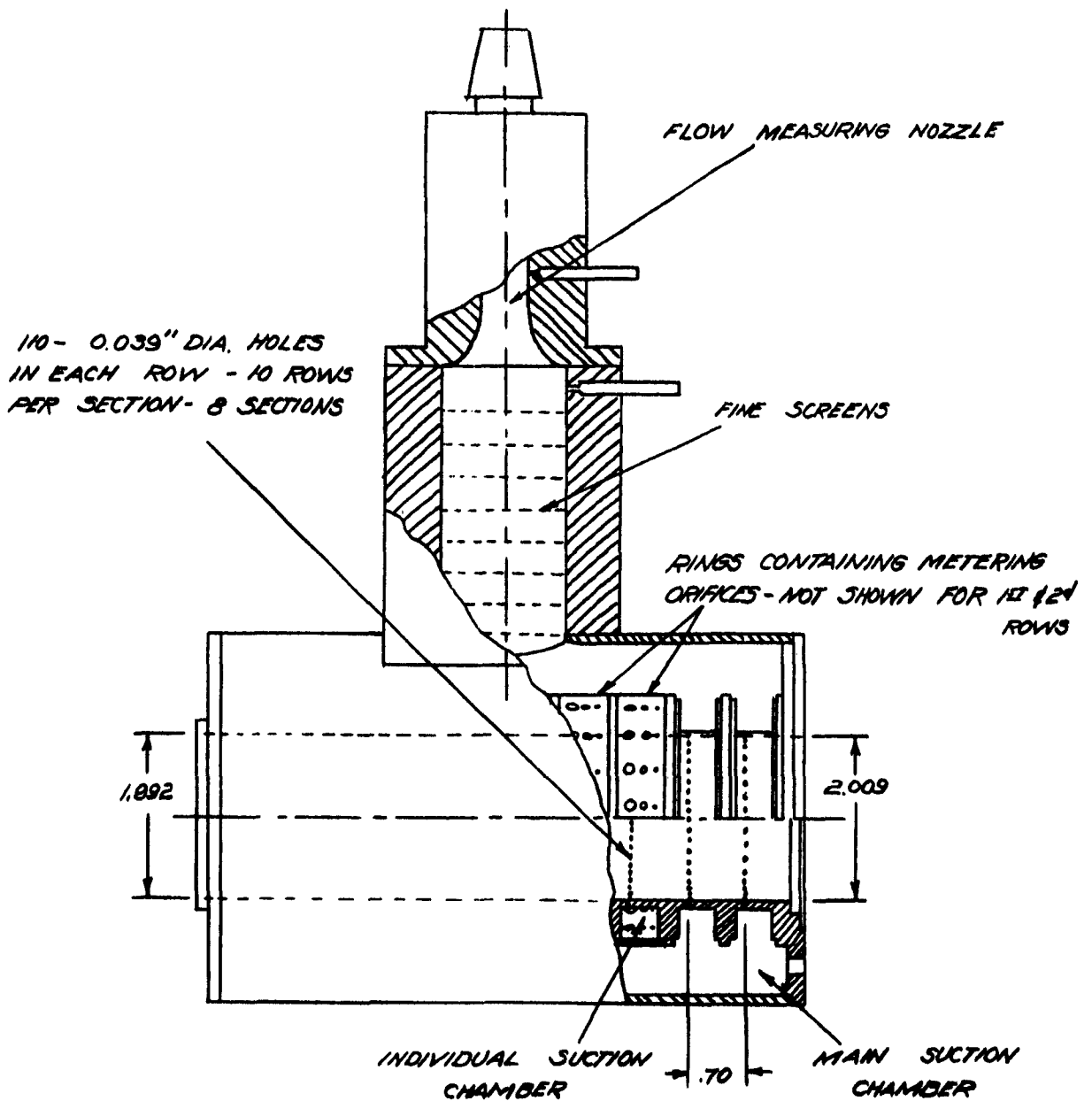
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								MODEL	
TABLE II COMPARISON OF 80 ROWS OF HOLES WITH 80 SLOTS 43% PRESS. RISE									
	HOLES	SLOTS	HOLES	SLOTS	HOLES	SLOTS	HOLES	SLOTS	
$\bar{U}$	73.18	72.81		68.57		87.63		97.6	93.2
$\bar{U}_1$	94.52	94.2		111.9		111.3		122.2	117.2
$\bar{U}_2$	71.0	71.7		83.7		84.2		91.37	88.2
$\bar{U}_3$	81.94	82.01		97.9		96.8		107.2	103.0
$\frac{V_{0.5} \times 10^{-6}}{D}$	15.6	15.6		18.8		18.5		20.6	19.4
$\theta - \text{FT.}$	.00312	.00238		.00273		.00188		.00259	.00168
$C_{D_{eff}} \times 10^4$	2.39	1.95		1.97		1.57		1.92	1.28
$C_{D_{eq}} \times 10^4$	5.05	4.89		4.94		4.59		4.93	4.72
$C_D \times 10^4$	7.43	6.84		6.91		6.10		6.85	6.00
PRESS. DIST.	LINEAR	CONVEX		LINEAR		CONVEX		LINEAR	CONVEX



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FIG. 2

DETAILS OF 1ST DIFFUSER  
SECTION

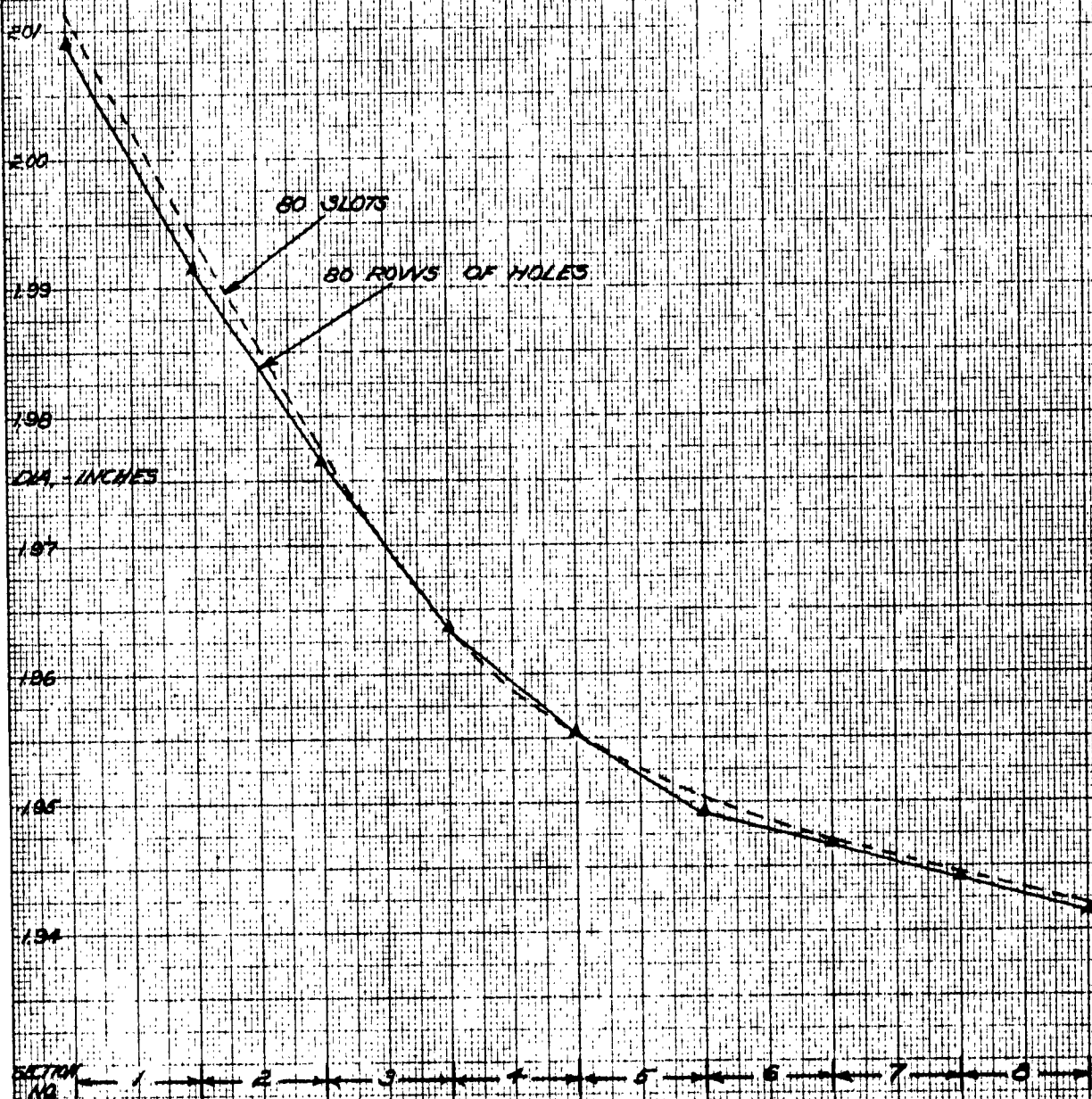


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FIG 3

VARIATION OF INSIDE DIAMETER ALONG  
SUCTION "DIFFUSER"



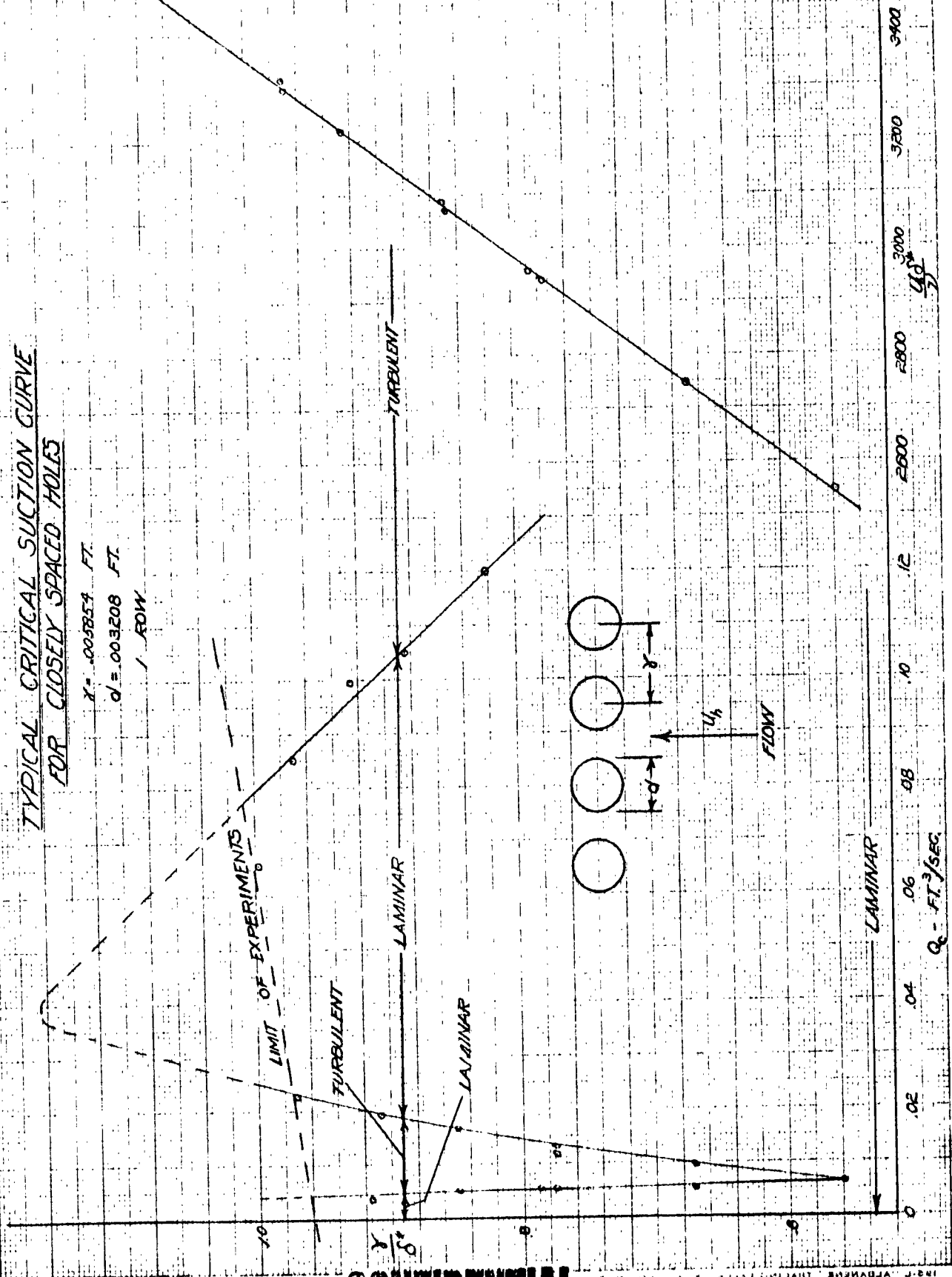
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FIG. 4

TYPICAL CRITICAL SUCTION CURVE  
FOR CLOSELY SPACED HOLES

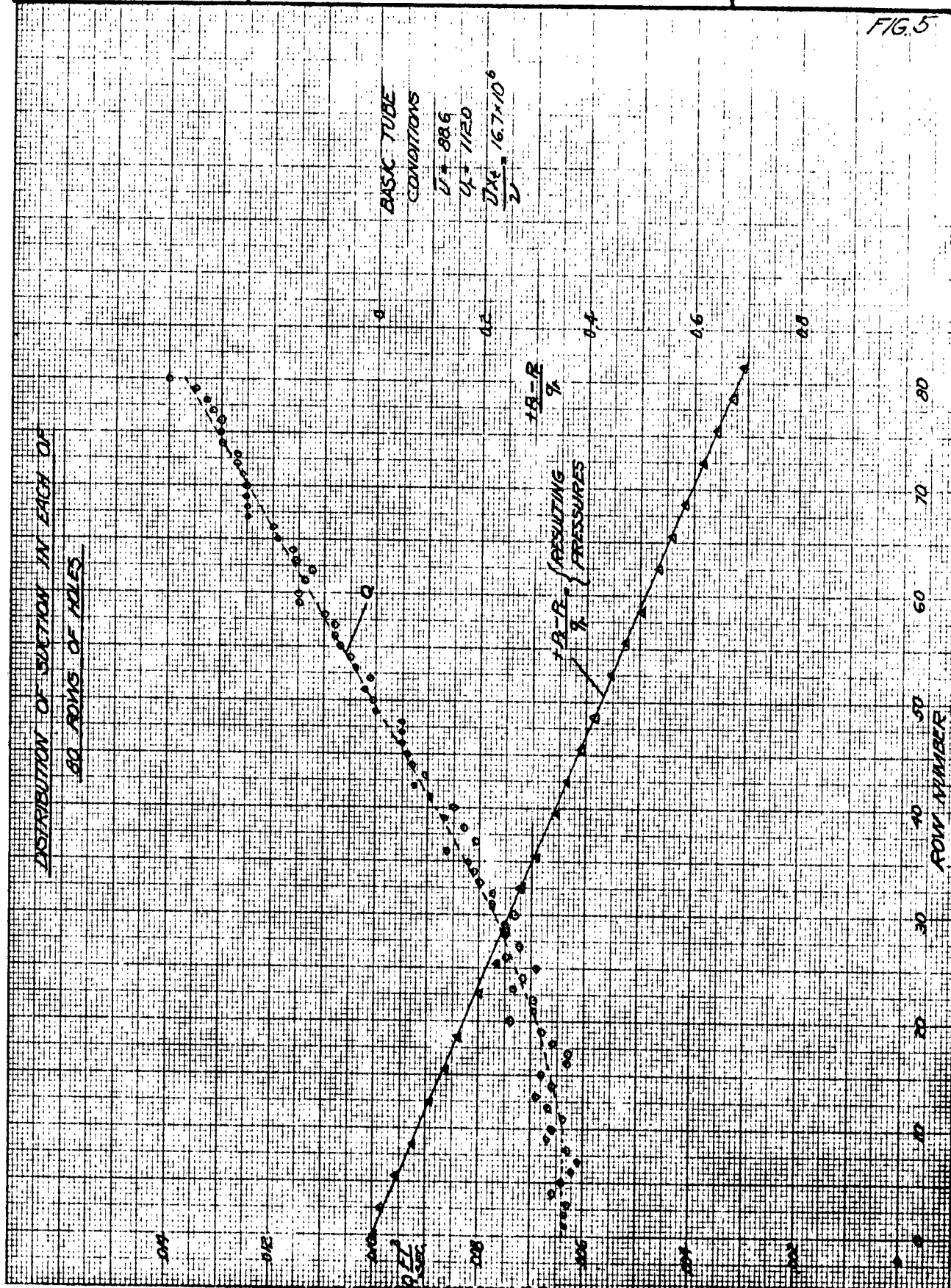
$\gamma = .003854$  FT.  
 $\alpha = .003208$  FT.  
/ ROW



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FIG. 5

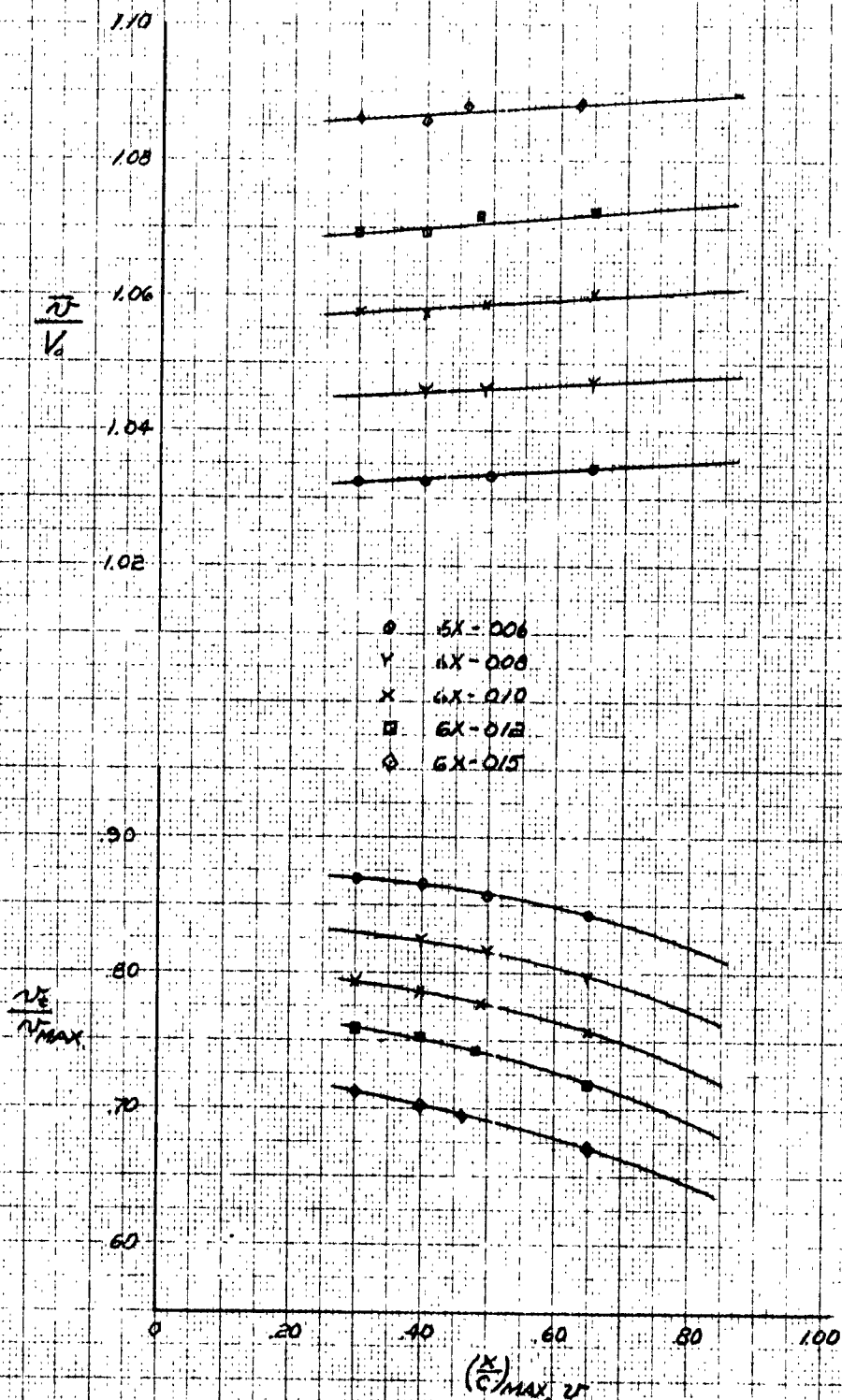


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TRENDS OF NACA G-SERIES  
AIRFOILS

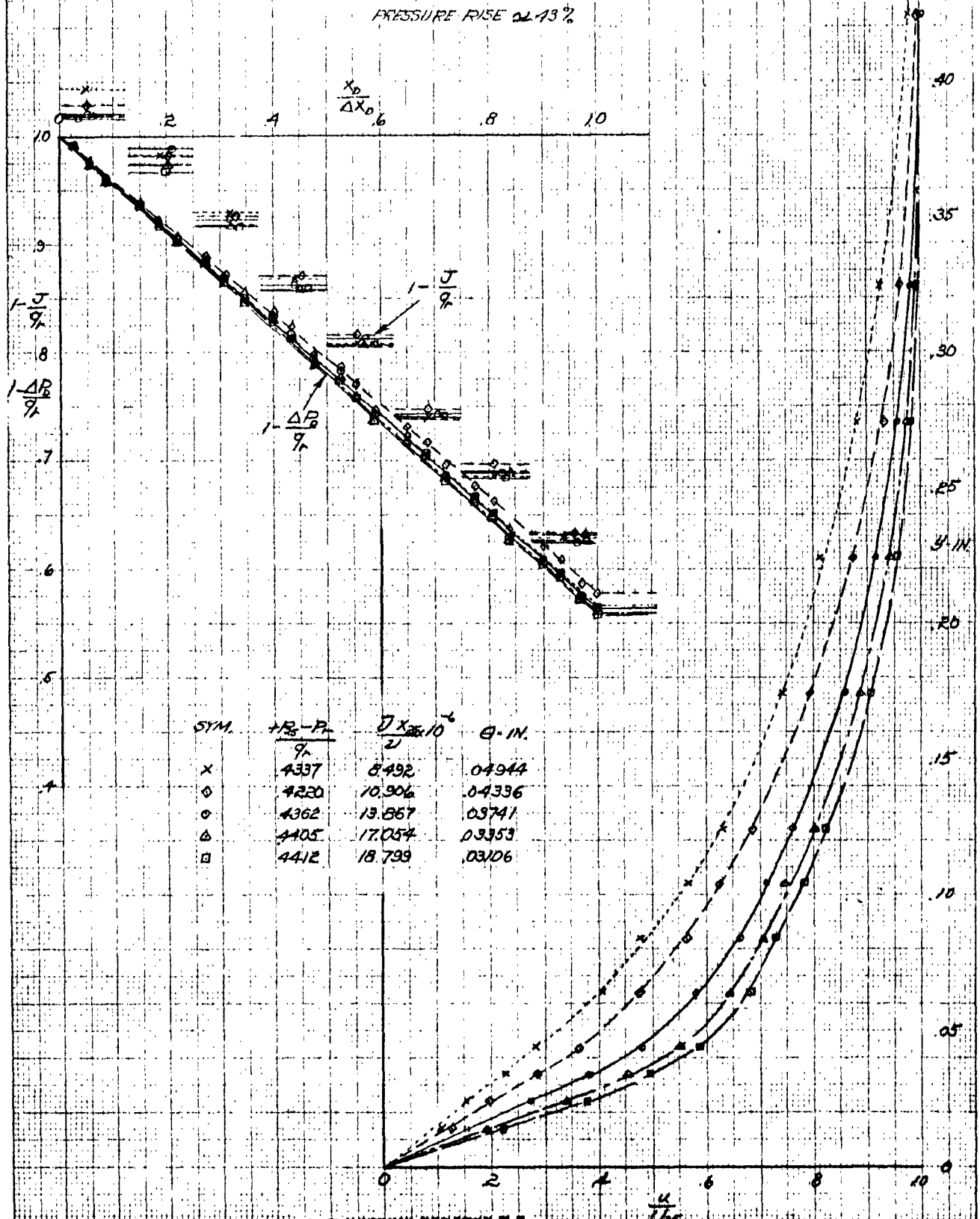
FIG. 6



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FIG 7

VARIATION OF DIFFUSER EXIT PROFILE WITH  
TUBE REYNOLDS NUMBER  
PRESSURE RISE 2.13%



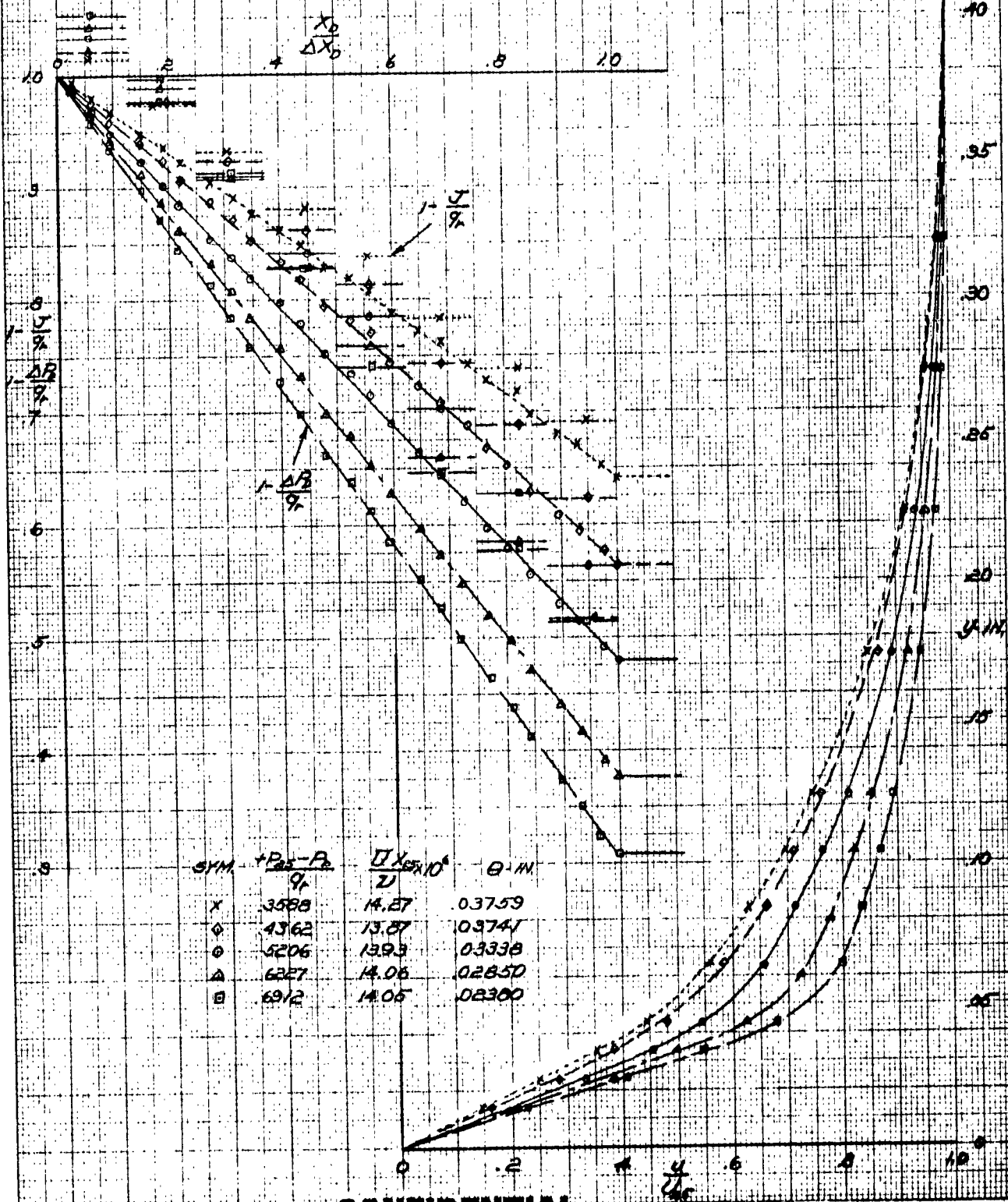
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FIG 8

VARIATION OF DIFFUSER EXIT PROFILE WITH  
DIFFUSER PRESSURE RISE

$$\frac{U_{\infty}}{U} \approx 14 \times 10^6$$

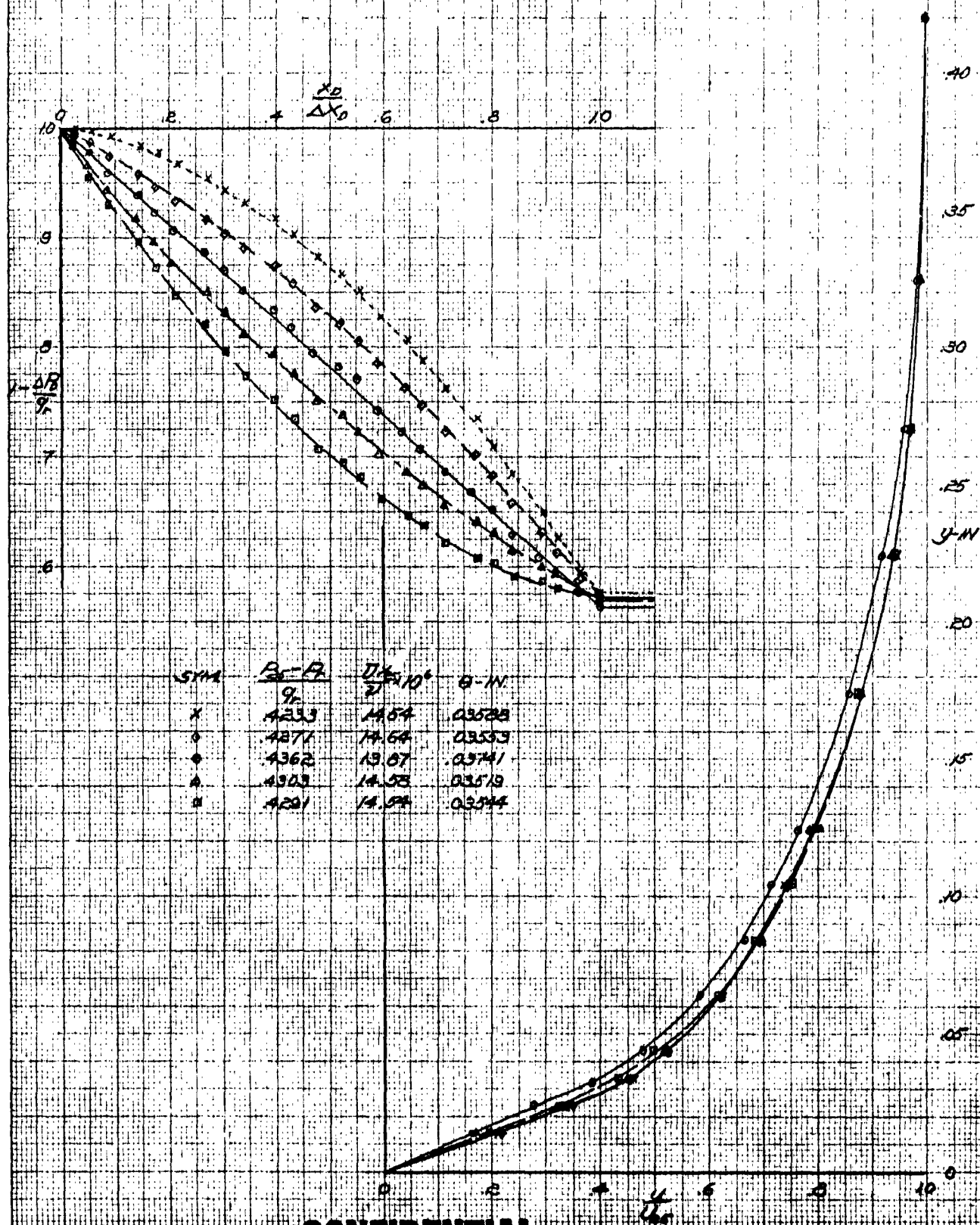


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FIG. 9

CONSTANCY OF DIFFUSER EXIT PROFILE WITH  
VARIATION OF DIFFUSER PRESSURE DISTRIBUTION



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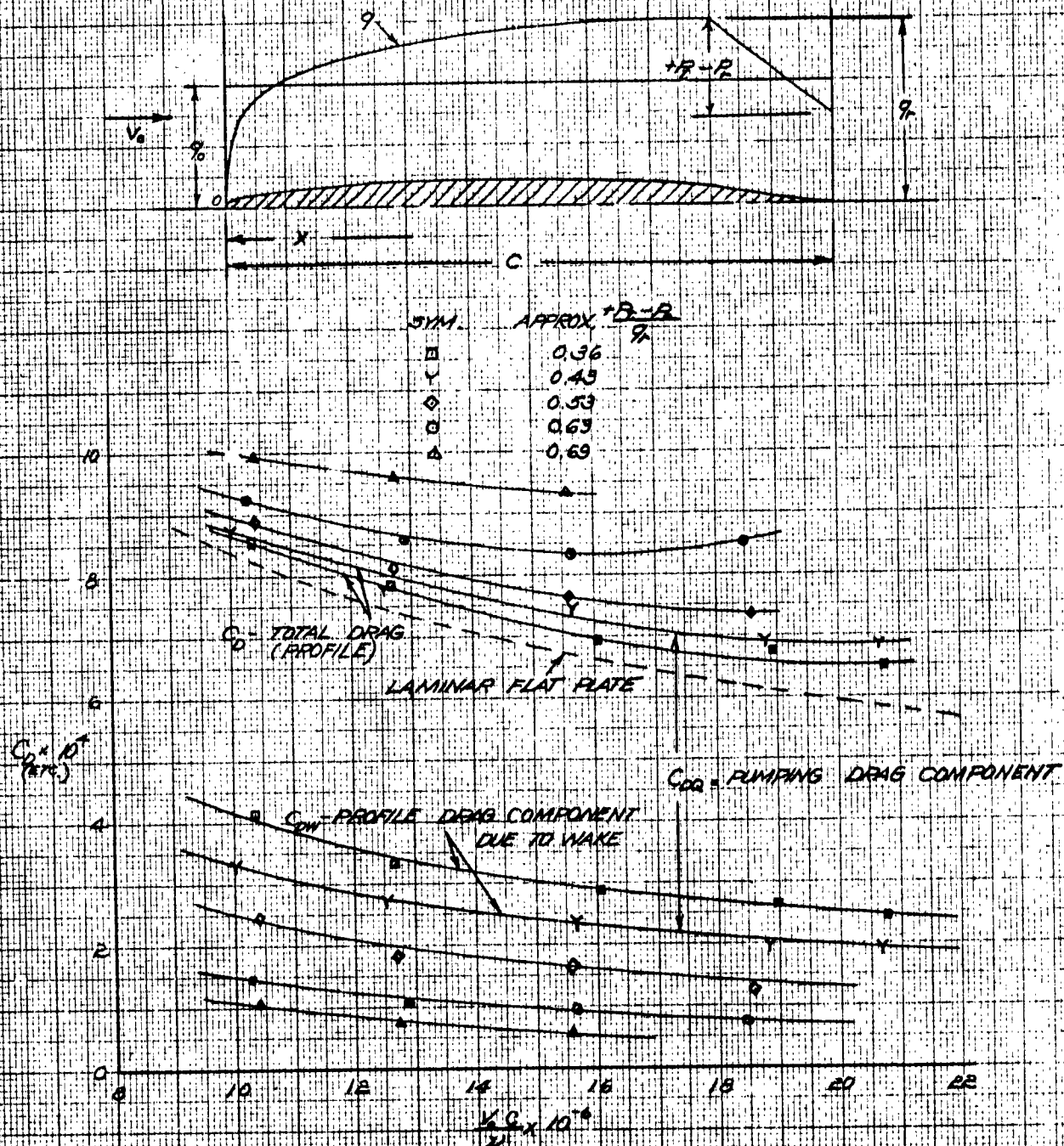
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MODEL

FIG. 10-A  
CONTINUED  
ON NEXT PAGE

DRA G COEFFICIENT OF EQUIVALENT  
AIRFOILS -  $\pi V_0 = 1.06$

INCLUDES UPPER & LOWER SURFACES  
C = 0



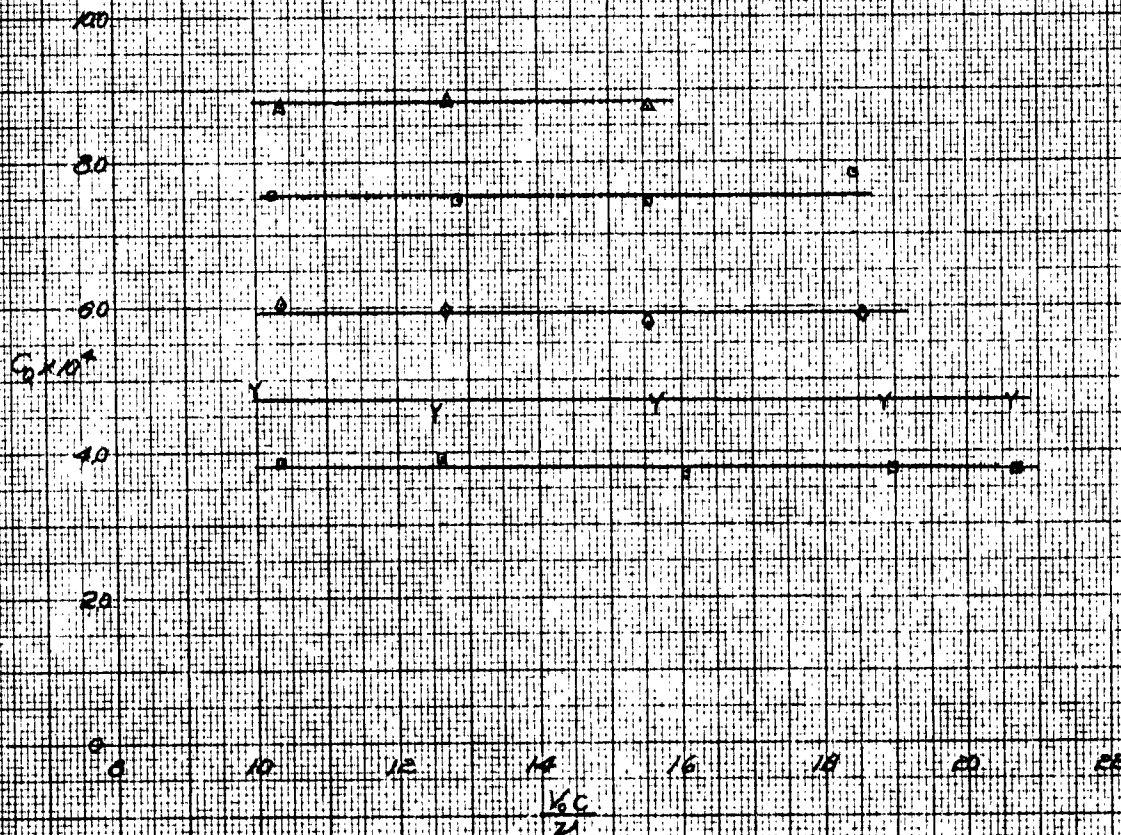
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FIG 10-B

VOLUME FLOW COEFFICIENT OF  
EQUIVALENT AIRFOILS

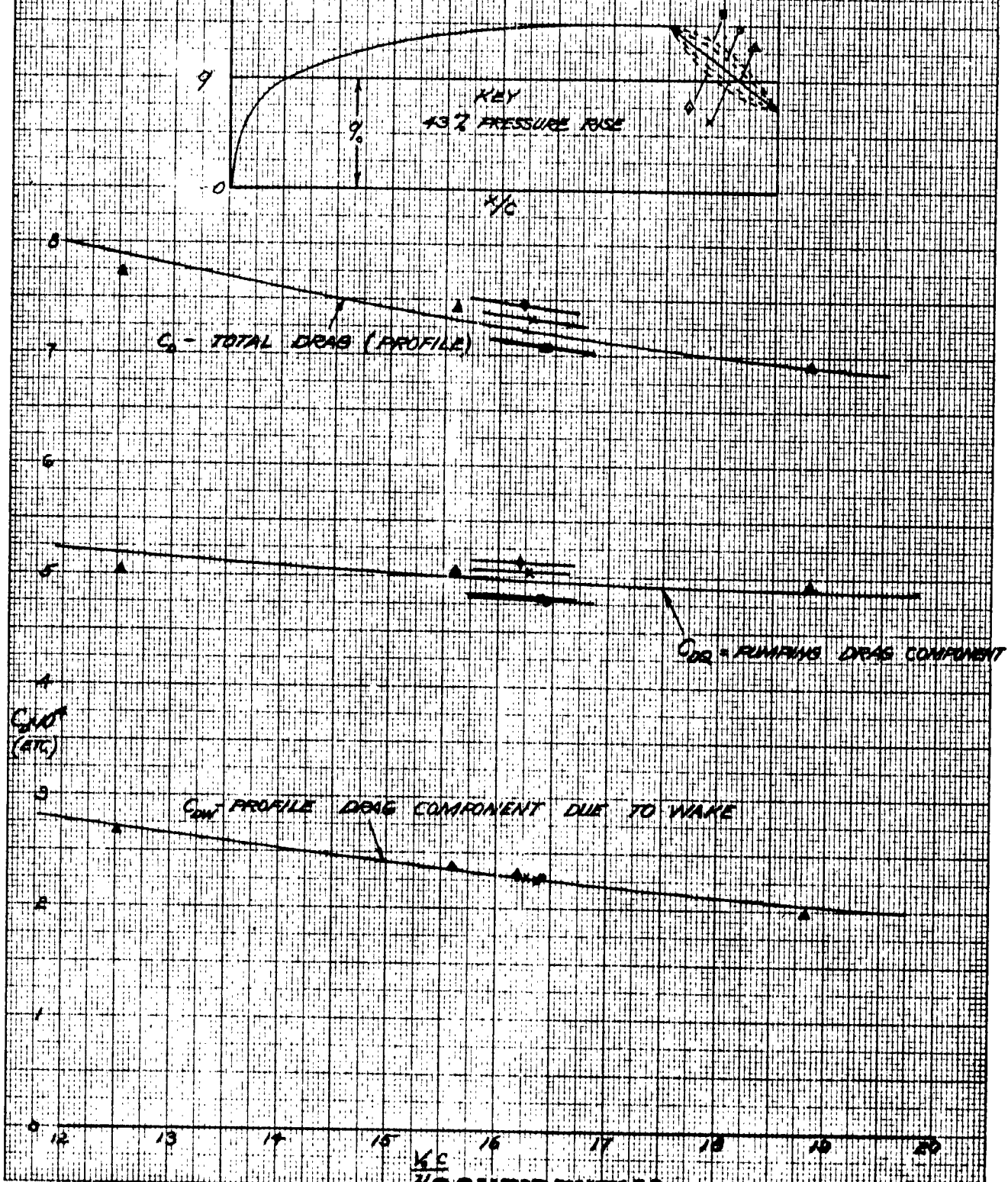
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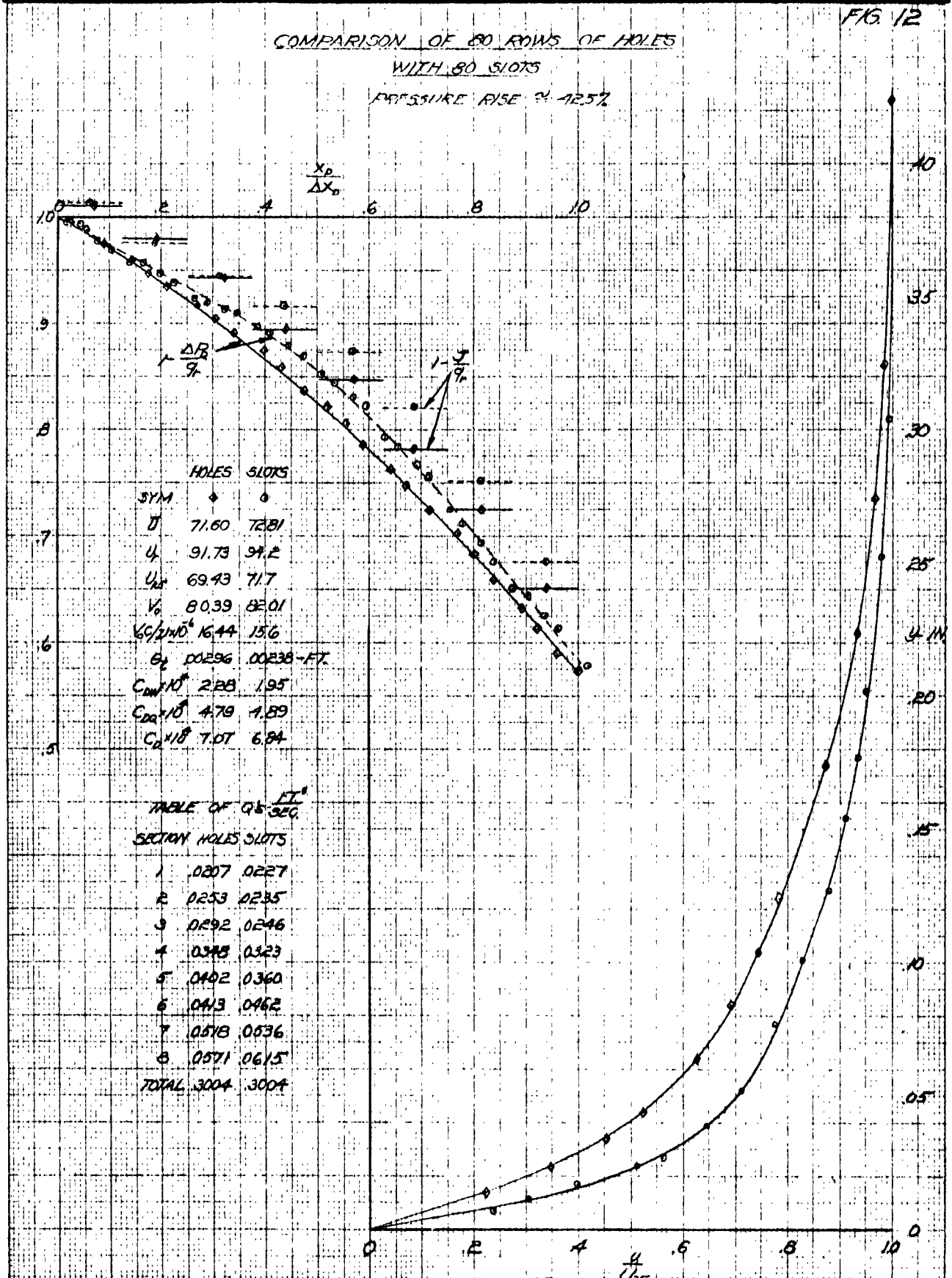
EFFECT OF TRAILING EDGE PRESSURE  
DISTRIBUTION ON AIRFOIL DRAG COMPONENTS  
INCLUDES UPPER & LOWER SURFACE -  $Q = 0$

FIG 11



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